C66-2772

[redacted]

4226-6010-RC000

APOLLO EXTENSION SYSTEMS TRAJECTORY STUDY

Contract No. NASw-1057 3 September 1965

[redacted]

DECL[redacted]

[redacted] 0

[redacted]

[redacted]

PREFACE

TRW Systems Group is conducting a set of studies under Contract No. NASw-1057 with the Office of Manned Space Flight, the National Aeronautics and Space Administration, for the evaluation and analysis of "advanced lunar missions trajectories". Generally, the tasks consist of the selection of representative trajectories and/or orbital techniques suitable for certain specified lunar and earth orbital missions. In the selection of such trajectories and orbits consideration is given to the effects of the variations of mission parameters and relevant constraints which are most limiting to mission performance. All of the specified missions are manned and involve the use of standard Apollo spacecraft or modifications thereof.

This document reports the results of such studies for a set of AES earth orbital and lunar missions. Representative flight trajectories were evaluated which tended to maximize the net disposable payload in orbit. Spacecraft ground track data was also determined. The launch vehicles were Saturn V and Saturn IB.

CONTENTS

Illustrations

Tables

Tables (Continued)

1.0 INTRODUCTION

The primary purpose of the study was to evolve and evaluate, in terms of net disposable payload, flight trajectory techniques for a typical set of AES missions which tended to optimize such disposable payload. Both booster and spacecraft propulsive capabilities were utilized to improve mission performance. The ground rules and constraints bounding the permissible flight techniques are specified in Section 4.0.

It is important to note that the study was essentially trajectory oriented, and as such, overall system considerations were not fully analyzed. On the other hand, it was a study objective to evolve meaningful and useful trajectory data. Consequently, wherever possible, system implications were estimated on the bases of experience and partial analysis. Examples of this type of consideration are the provision for manned earth entry within CM design characteristics and the establishment of a preliminary range safety model for ETR launch. An example of a system feature which could not be considered, since mission analysis and definition in depth was necessary, is the spacecraft configuration for the several missions, from the point of view of weight, location, and usage of all of the mission expendables.

2.0 SUMMARY

The performance capabilities of the Saturn launch vehicles have been evaluated for AES mission planning purposes. Reference trajectories were selected for representative missions and are included in this report.

A summary of the net payload capabilities for the Saturn IB launch vehicle is given in Table 2-1. A payload capability summary for the Saturn V Earth Orbital Mission is given in Table 2-2. The net or discretionary payload is defined herein as the total weight on orbit less the jettison weight of the spent S-IVB stage used for final injection less crew/mission expendables less CSM deboost requirements. In some cases the first burn of the Service Module established the mission orbit. For these cases the net payload refers to the difference between SM burnout weight capability and the sum of crew expendable and deboost requirements.

Table 2-1. Saturn IB Performance Summary (200 n mi circular orbits, 45 day mission)

	Flight Mode	Launch Azimuth (Deg)	Orbit Inclination (Deg)	Gross CSM Weight a) on Orbit	Net Payload
1.	Direct ascent, injection by the S-IVB	90	28.5	26,050	-
2.	Hohmann transfer from an 80 n mi parking orbit. Transfer maneuver provided by two impulsive burns of the CSM	90	28.5	33,185	5,595
3.	Hohmann transfer from an 80 n mi parking orbit. Transfer maneuver provided by two impulsive burns of the CSM	44	50.3	30,930	3,340
4.	Direct ascent, un-manned payload	90	28.5	N.A.	26,723
5.	Hohmann transfer from an 80 n mi parking orbit. Transfer maneuver provided by two impulsive burns of the CSM	182 b)	90	25,419	-
6.	Hohmann transfer from an 80 n mi parking orbit. Transfer maneuver provided by two impulsive burns of the CSM	189.4 b)	96.5	24,568	-
7.	Suborbital start of CSM, Hohmann transfer from an 80 n mi perigee. CSM restarts for apogee injection.	182 b)	90	28,425	835
8.	Suborbital start of CSM, Hohmann transfer from an 80 n mi parking orbit. CSM restarts for apogee injection. Yaw program of one deg/sec between 110 and 134 and between 146 and 164.1 sec provides the polar inclination	146 c)	90	24,375	-

a) Includes 5310 lbs of crew/mission expendables and 1080 lbs of SM [illegible] for deboost (perigee at earth's surface), [illegible], and [illegible]% contingency factor. CSM weight required for 45 day mission and deboost is 27,590 lbs

Table 2-2. Saturn V Performance Summary (Specified 45-day missions) for 200 n.mi and Synchronous Altitudes

	Flight Mode	Launch Azimuth (Deg)	Orbit Inclination (Deg)	Gross CSM Weight on Orbit	a), b) Net Payload
1.	Direct ascent	90	28.5	232,955	205,365
2.	Direct ascent	44	50.3	222,650	195,060
3.	Direct ascent	114	36.4	229,670	202,080
4.	Direct ascent	146	59.4	217,120	189,530
5.	Direct ascent, yaw program of one deg/sec from 392 to 480.3 sec.	44	90	49,940	22,350
6.	Direct ascent, yaw program of one deg/sec from 163 to 250.1 sec (S-IVB stage was removed)	114	90	86,628	59,038
7.	Direct ascent, yaw program of one deg/sec from 120 to 150 and between 159 and 172.4 sec, S-IC cutoff 4 sec early	146 c)	90	157,380	129,790
8.	Direct ascent, yaw program of one deg/sec from 120 to 150 and between 159 and 181.9, S-IC cutoff 4 sec early	146 c)	96.5	145,253	117,663
9.	Synchronous orbit, first S-IVB burn establishes a transfer orbit to synchronous altitude. The remaining S-IVB propellant is consumed at apogee and establishes an intermediate orbit for transposition and docking. At the next apogee CSM ignites to circularize orbit saving enough propellant for deboost to Apollo landing site	90	28.5	69,689	27,929
10.	Synchronous orbit, first S-IVB burn establishes a 100 n mi parking orbit for coast to the equator. There it restarts to provide transfer ellipse and a two-deg plane change. The third burn, at apogee, establishes an intermediate orbit for transposition and docking. Fully loaded CSM ignites at the next apogee finishing the plane change and circularizing the orbit. Deboost is to an equatorial landing site.	90	0	61,334	19,574

3.0 VEHICLE CHARACTERISTICS

The launch vehicle and CSM characteristics used for performance analyses and trajectory simulation are given in this section.

Service Module propulsion was applied impulsively, the single exception being the Saturn IB earth polar mission in which the Service Module propulsion was utilized in a sub-orbital start. It was assumed in this case that the payload would be contained within the SM and would be deboosted from orbit with the CSM.

3.1 Launch Vehicles

3.1.1 Saturn IB

The Saturn IB configuration data are summarized in Tables 3-1 and 3-2. The S-IVB jettison weight breakdown is given in Table 3-3 for both Saturn IB and Saturn V application. The data were obtained from Reference 1.

3.1.2 Saturn V

Table 3.4 lists the Saturn V data used. The axial force coefficient (from Reference 2), given as a function of Mach Number in Figure 3-1, was used on all Saturn V configurations. The data were obtained from Reference 3.

3.2 Spacecraft

The Block II Apollo spacecraft characteristics are given in Table 3-5. Control weights as specified in Reference 4 were used.

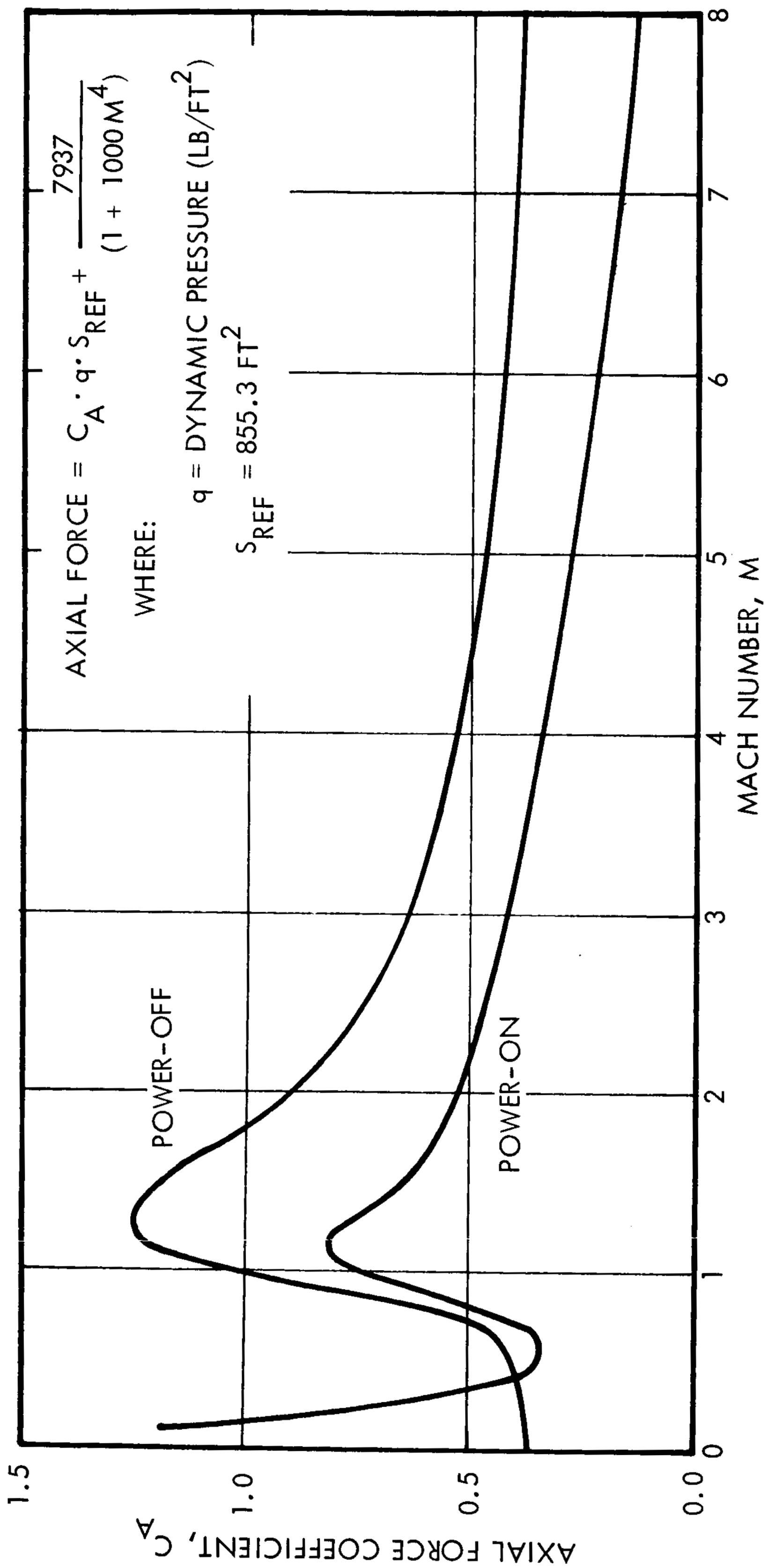


FIGURE 3.1 SATURN V AXIAL FORCE COEFFICIENT

Table 3-1. Saturn IB Configuration Data Summary

Stage	S-IB	S-IB	Coast	S-IVB	S-IVB	S-IVB	Coast	SM
Phase	1	2	3	4	5	6	7	8
Vacuum Thrust, (lb)	Table	Table	-	205,000	230,000	190,000	-	21,800
Vacuum Specific Impulse (sec)	-	-	-	426	423.1	427.47	-	313
Weight Decay Rate, (lb/sec)	Table	Table	-	481.220	543.606	444.475	-	69.65
Usable Propellant 1), (lb)	864,461	21,873	-	4,812	154,928	68,716	-	41,000
Jettison Weight, (lb)	-	105,826	-	235	8,000 2)	34,985 3)	-	10,200
Mixture Ratio	-	-	-	5.0	5.5	4.7	-	-
Phase Duration, (sec)	134.72	6.92	4.30	10.00	285.00	154.60	5.40	589.00

1) The S-IVB and SM were offloaded in some cases

2) Jettison 171.75 sec from liftoff

3) A breakdown is given in Table 3-~~5~~ 3

Table 3-2. Saturn S-IB Propulsion and Aerodynamic Data

Time (sec)	Thrust (lb)	Weight Decay Rate (lb/sec)	Dynamic Pressure (lb/ft^2)	Drag (lb)	Weight (lb)
0.0	1,672,085	6416.0636	0.00	0.00	1,298,819
10.0	1,690,237	6438.6823	12.60	4851.95	1,234,545
20.0	1,707,931	6458.2350	60.20	18,205.60	1,170,061
40.0	1,751,190	6462.8884	291.49	52,355.19	1,040,796
60.0	1,808,340	6446.4649	608.54	162,856.31	911.715
80.0	1,854,578	6444.3372	644.36	144,292.33	782,760
100.0	1,874,714	6387.7477	286.73	42,872.65	654,446
120.0	1,876,528	6340.6920	75.52	9,475.15	527,129
134.72	1,846,736	6265.6196	23.09	2896.93	434,345
134.72	917,333	3158.6360	23.09	2896.93	434,345
141.64	917,333	3158.6360	11.0	1,380.53	412,472
141.64	0	0	11.0	1,380.53	306,646

Table 3-3. S-IVB Jettison Weight

A. For Saturn IB Application

Dry Weight at Injection	23,586	LB
Residuals and Reserve	1,949	
Flight Performance Reserve	1,500	
Instrumentation Unit	4,150	
LEM Adapter	3,800	
Jettison Weight	34,985	LB

B. For Saturn V Application

Dry Weight at Injection	27,400	LB
Residuals and Reserve	2,305	
Flight Performance Reserve	2,884	
Instrumentation Unit	4,150	
LEM Adapter	3,800	
Jettison Weight	40,539	LB

Table 3-4. Saturn V Data Summary

Stage	S-IC	S-IC	Coast	S-II	S-II	S-II	Coast	S-IVB	SM
Phase	1	1	3	4	5	6	7	8	9
Vacuum Thrust, (lb)	8,745,395	6,996,316	-	1,035,000	1,135,000	960,000	-	207,000	21,800
Sea Level Thrust, (lb)	7,610,000	6,088,000	-	-	-	-	-	-	-
Vacuum Specific Impulse, (sec)	302.90	302.90	-	426.00	423.57	427.57	-	426.00	313
Sea Level Specific Impulse, (sec)	263.58	263.58	-	-	-	-	-	-	-
Usable Propellant (lb)	4,462,618	92,389	-	24,296	573,608	332,088	-	230,972	41,000
Weight Decay Rate, (lb/sec)	28,871.69	23,097.35	-	2,429.60	2,681.50	2,245.25	-	485.92	69.65
Jettison Weight, (lb)	0	381,645	-	0	17,970*	100,921	-	40,539**	10,200
Phase Duration, (sec)	154.57	4.00	3.80	10.00	213.91	147.91	4.80	475.33	589

Phase 2 represents conditions after center-engine shutdown.
Phases 4, 5, and 6 represent S-II programmed mixture ratio variations.

*S-IC interstage weight of 9770 lb jettison at t = 188.57 sec.
Launch escape system (LES) of 8200 lb jettisoned at t = 193.57 sec.

**Includes 2884 lb propellant margin for performance pad and LEM adapter.

Table 3-5. Block II Spacecraft Definition

Command Module Weight (with crew)	11,000 Lbs
Service Module	
Injected Inert Weight	10,200 Lbs
Propellant Tank Capacity	41,000 Lbs
I_{SP}	313 Sec
Thrust	21,800 Lbs
LEM Descent Stage	
Propellant Tank Capacity	15,920 Lbs
I_{SP}	302 Sec
Thrust (Max)	10,500 Lbs
LEM Ascent Stage	
Propellant Tank Capacity	4,920 Lbs
I_{SP}	303 Sec
Thrust	3,500 Lbs

4.0 GROUND RULES AND CONSTRAINTS

Some of the more pertinent ground rules and constraints used to conduct the performance calculations are given below.

Since very significant payload penalties were involved, preliminary investigations were conducted to delineate a range safety model to constrain the AES trajectories. Normal operational procedure is for the Contractor requesting a Range Safety Waiver to prepare a Hazard Report. This is a detailed report containing the nominal proposed trajectory with the probable dispersed trajectories resulting from off-nominal vehicle performance, probable failure modes, and probabilities of land impact and population kill. After a review by Range Safety with the importance of the mission weighted against the relative danger, the use of the range will be granted or denied. For the preliminary trajectory investigations of this study, the model below was considered reasonable and practical and commensurate with the scope of the study.

a) Range Safety Model

   1) An IIP overflight of the Continental United States or Canada is not permissible at suborbital speeds. This is defined as requiring that the three-sigma right or left deviating trajectory to be a minimum of 25 miles from the affected coast line.

   2) All planned stage impact points shall be in the ocean area. The determination of such planned impact points involves an assessment of the impact dispersion ellipse. No land shall be within 25 miles of the ellipse.

b) Launch will be from Cape Kennedy

c) Launch vehicle and CSM characteristics are as summarized in Section 3.0.

d) All missions are manned (rendezvous vehicle may not be).

   1) External spacecraft configurations are similar to that of Apollo.

   2) Deboost from orbit must be compatible with CM re-entry design conditions.

   3) Recoveries are generally planned for either of the two Apollo impact areas.

e) The trajectories are to be shaped for near maximum payload within the established range restrictions. When payloads exceed the 110,000 pound stack limit of S-IVB, alternate flight modes will be offered abiding by the stack limit.

f) The upper limit on booster turning rates is one degree per second.

g) For performance contingencies of the Service Module, an additional ten percent is to be provided with each velocity increment plus a 100 fps increment for a return midcourse maneuver.

h) The CSM/LEM docking maneuver is to be alloted a time of twenty minutes.

i) Although the S-IVB propellant ~~burn~~ loading history will be different than that of Apollo, it will be assumed that its propellant pressurization subsystem will be modified to permit two or three burns with a propellant loading history commensurate with the intended mission.

5.0 MISSION PROFILES

The results of this study are given in this section. Performance, reference trajectory, vehicle and propulsive history, and ground track data are shown for the various AES earth orbital, lunar orbital and lunar exploration missions. Since the intent of the study was to optimize performance for the several missions, particularly the earth orbital missions, a variety of vehicle ascent and spacecraft injection techniques were investigated, utilizing Hohmann transfer from low earth orbit and Service Module propulsion, as applicable.

Table 5-1 indicates the Saturn IB low altitude earth orbit mission cases for which performance and trajectory data were calculated. Note that several of the cases involve a suborbital start of the Service Module. Such a start implies that the net discretionary payload (payload available for the mission experiments) cannot be carried within the LEM adapter, inside of a LEM or otherwise, but must be contained within the CSM. The intent of the 182° azimuth launches is to place a performance outer bound on the vehicle capabilities to accommodate variations in the maximum launch azimuth permitted by range safety constraints.

Table 5-2 indicates the Saturn V low earth and synchronous altitude orbit mission cases for which performance and trajectory data were derived. The lunar orbital and exploration missions have been previously reported in references (5) and (6). The particular emphasis of the studies reported in these references were on the generation of lunar orbital and translunar and transearth injection techniques which maximized the lunar accessibility for the several mission objectives.

Synchronous orbit studies also included descent and deboost alternatives to equatorial and off-equatorial (Apollo landing areas) landings involving LEM stage propulsion. These cases are discussed in Section 5.2. Table 5-3 indicates the missions and cases for which reference trajectories were derived. Lunar missions are also included, launch dates and pertinent lunar orbital and surface objectives being arbitrarily stipulated.

Table 5-1. Saturn IB Missions

	Flight Profile	Orbit Inclination (Deg)	Circular Orbit Altitude (n.mi.)
1.	Direct ascent, initial launch azimuth = 72°, no CSM burn (computed for configuration simulation comparison)	32.6	105
2.	Direct ascent, azimuth = 90° (Hohmann transfer to 200 n.mi. with two impulsive burns of the CSM)	28.5	80
3.	Direct ascent, azimuth = 182° (Hohmann transfer to 200 n.mi. with two impulsive burns of the CSM)	90	80
4.	Direct ascent, azimuth = 189°.4	96.5	80
5.	Direct ascent, azimuth = 182°	90	100
6.	Direct ascent, azimuth =182°	90	200
7.	Hohmann transfer from 80 n.mi. perigee, azimuth = 90°, perigee overspeed provided by S-IVB, injection at apogee by CSM	28.5	200
8.	Same as case 7 but azimuth = 182°	90	200
9.	Direct ascent, azimuth = 146°, yaw program of one deg/sec initiated at S-IVB ignition (Hohmann transfer to 200 n.mi. with two impulsive burns of the CSM)	90	80
10.	Same as case 9 but yaws for a longer duration	96.5	80
11.	Hohmann transfer from 80 n.mi. perigee, azimuth = 146°, CSM burned as a third stage (no docking maneuver). CSM restarts for injection at apogee. Yaw program during S-IB and S-IVB operation.	90	200
12.	Same as case 11 but S-IVB propellant was offloaded for maximum payload.	90	200
13.	Hohmann transfer from 80 n.mi. perigee, azimuth = 182°, CSM burned as a third stage (no docking maneuver). CSM restarts for injection at apogee.	90	200

Table 5-1. Saturn IB Missions (cont'd)

	Flight Profile	Orbit Inclination (Deg)	Circular Orbit Altitude (n.mi.)
14.	Same as case 13 but S-IVB propellant was offloaded for maximum payload	90	200
15.	Unmanned Saturn IB, direct ascent, azimuth = 182°	90	200
16.	Same as 15 but azimuth = 90°	28.5	200
17.	Unmanned Saturn IB, direct ascent, azimuth = 90°	28.5	80

Table 5-2. Saturn V Earth Orbital Missions

Flight Profile	Orbit Inclination (Deg)	Circular Orbit Altitude (n.mi.)
1. Direct ascent, initial launch azimuth = 90°, single burn of S-IVB	28.5	200
2. Same as case 1 but azimuth = 44°	50.2	200
3. Direct ascent, azimuth = 44°, yaw (north) program of 1 deg/sec initiated 392 sec from launch	90	200
4. Same as case 1 but azimuth = 114	36.4	200
5. Direct ascent, azimuth = 114, yaw (south) program initiated 400 sec from launch (spent S-II impacts in South America)	90	200
6. Direct ascent, azimuth = 146, yaw program initiated at S-II start (jettisoned LES impacts in Cuba)	90	200
7. Same as 6 but LES impact adjusted for open water impact by early S-IVB cutoff	90	200
8. Direct ascent, azimuth = 146, yaw program initiated at S-II start (jettisoned LES impacts in Cuba)	96.5	200(sun synchronous)
9. Same as 8 but LES impact adjusted for open water impact by early S-IVB cutoff	96.5	200
10. Same as case 7 but payload constrained to S-IVB stack limit of 110,000 lb	90	200
11. Same as case 9 but payload constrained to S-IVB stack limit of 110,000 lb	96.5	200
12. Direct ascent, azimuth = 146°, no yaw		200
The following cases were for a two-stage arrangement with the S-IVB removed.		
13. Direct ascent, azimuth = 114°, no yaw	36.4	200

Table 5-2. Saturn V Earth Orbital Missions (cont'd)

	Flight Profile	Orbit Inclination (Deg)	Circular Orbit Altitude (n.mi.)
14.	Same as case 13 but yaw (south) program initiated at S-II start	90	200
15.	Same as case 14 but azimuth = 146°	90	200
16.	Same as case 13 but yawed for longer duration	96.5	200
17.	Same as case 15 but yawed for long duration	96.5	200
	The following cases were for the purpose of determining the synchronous orbit payload capabilities. All launches were due east from ETR.		
18.	Three-burn S-IVB, first to establish a 100 n.mi. parking orbit, restart at equator for perigee overspeed and third burn for injection at apogee. CSM was offload to the propellant required for deboost.	28.5	19,329
19.	Same as case 18 except CSM was fully loaded. The third S-IVB burn (at apogee) an intermediate elliptical parking orbit is established for transposition and docking. At the next apogee CSM starts and accelerates to circular velocity saving enough propellant for deboost.	28.5	19,329
20.	Same as case 18 except orbit is turned equatorial during third S-IVB operation	0	19,329
21.	Same as case 19 except orbit is turned equatorial during S-IVB and CSM operation	0	19,329
22.	Single-burn S-IVB, S-II injects offloaded S-IVB onto 100 n.mi. parking orbit. S-IVB starts at equator and changes orbit plane 15 deg and simultaneously provides perigee overspeed. Fully loaded CSM ignites at apogee and turns the orbit plane equatorial and circularizes orbit. (Payload capability was marginal).	0	19,329

Table 5-2. Saturn V Earth Orbital Missions (cont'd)

	Flight Profile	Orbit Inclination (Deg)	Circular Orbit Altitude (nmi.)
23.	Two-burn S-IVB, S-II injects offloaded S-IVB onto 100 n.mi. parking orbit. S-IVB starts at equator and establishes perigee. S-IVB restarts at apogee for circularization. CSM is offloaded to the deboost requirement.	28.5	19,329
24.	Two-burn S-IVB, S-II injects offloaded S-IVB onto 100 n.mi. parking orbit. S-IVB starts at equator and changes orbit plane 2 deg and establishes perigee. S-IVB restarts at apogee and provides an intermediate parking orbit for docking. At a subsequent node fully loaded CSM ignites, finishes the plane change and circularization saving enough propellant for deboost.	0	19,329
25.	Three-burn S-IVB, first to establish parking orbit, restart at equator changing the orbit plane 2 deg and providing perigee overspeed, restart at apogee establishing intermediate parking orbit. Fully loaded CSM ignites at the next apogee finishing the orbit but saving enough propellant for deboost.	0	19,329
26.	Same as 25 but no plane change.	28.5	19,329
27.	Same as 25 except all CSM propellant is consumed during apogee maneuver. Deboost was provided by an offloaded LEM descent stage.	0	19,329
28.	Same as 27 but no plane change.	28.5	19,329

Table 5-3. AES Reference Trajectories

1. Saturn IB reference trajectories; 200 n.mi. circular orbit by Hohmann Transfer (perigee = 80 n.mi., i = 28.5 deg)
2. Descent from 200 n.mi. orbit; i = 28.5 deg.
3. Saturn IB reference trajectories; 200 n.mi. circular orbit by Hohmann Transfer (perigee = 80 n.mi., i = 50.3 deg)
4. Descent from 200 n.mi. orbit; i = 50.3 deg
5. Saturn IB reference trajectories; 200 n.mi. circular orbit by Hohmann Transfer (perigee = 80 n.mi., i = 90 deg)
6. Descent from 200 n.mi. orbit; i = 90 deg
7. Saturn IB reference trajectories; 200 n.mi. circular orbit by Hohmann Transfer (perigee = 80 n.mi., i = 96.5 deg)
8. Descent from 200 n.mi. orbit; i = 96.5 deg
9. Saturn V reference trajectories, low-earth orbits (200 n.mi. circular, i = 28.5 deg)
10. Saturn V reference trajectories, low-earth orbits (200 n.mi. circular, i = 90 deg)
11. Saturn V reference trajectories, low-earth orbits (200 n.mi. circular, i = 90 deg, S-IVB stack limit)
12. Saturn V reference trajectories, low-earth orbits (200 n.mi. circular, i = 96.5 deg)
13. Saturn V reference trajectories, low-earth orbits (200 n.mi. circular, i = 96.5 deg, S-IVB stack limit)
14. Saturn V reference trajectories, synchronous orbits (i = 28.5 deg)
15. Descent from synchronous orbit; i = 28.5 deg
16. Saturn V reference trajectories, synchronous orbits (i = 0 deg)
17. Descent (on equator) from synchronous orbit; i = 0 deg
18. Saturn V lunar exploration mission, translunar trajectory
19. Saturn V lunar exploration on mission, unmanned LEM

Table 5-3. AES Reference Trajectories (cont'd)

20. Saturn V lunar exploration mission, manned LEM

21. Transearth free-flight; 109.17 hours

22. Scientific survey, 28-day lunar orbit

23. Saturn IB (suborbital start of Service Module) reference trajectories; 200 n.mi. orbit by Hohmann transfer, yaw program during boost (i = 90)

In general, integrated trajectories were computed for the ascent phases into parking orbit and the descent phase from final orbit to an earth altitude of 400,000 feet. Intermediate maneuvers were computed on an impulsive velocity bases. Nominal landing points were calculated without considerations of the aerodynamics of the entry body.

5.1 Low Altitude Earth Orbits

The data herein includes performance and trajectory calculations for both Saturn V and Saturn IB booster vehicles and for low inclinations and high inclinations at altitudes to 200 n.mi.

5.1.1 Low Inclination Orbits

5.1.1.1 Saturn IB Launch Vehicle

The direct ascent burnout weight capability of the Saturn IB into a 200 n.mi. circular orbit launched due east from ETR ($28^{\circ}.5$ orbital inclination) is 61,050 pounds. The jettison weight of the S-IVB as given in Table 3-3 is 34,985 pounds including the 3800 pound LEM adapter. After jettisoning the S-IVB and performing the transposition and docking maneuver the total weight remaining is 26,065 pounds. To determine the weight that might be available for experiments, the crew expendables for a 45-day mission and the CSM weight required for descent was computed and compared with the total payload (26,065). Table 5-4 gives an estimate of the additional crew/mission expendables required for a 45-day mission. The CSM weight required for deboost from a 200 n.mi. circular orbit is 22,280 pounds. The breakdown of this weight is given in Table 5-5. Thus, the weight required for this particular 45-day mission is 27,590 pounds (22,280 + 5,310). This is greater than the capability of the launch vehicle, and thus a direct ascent mission of this nature is not possible.

To increase the mission performance the ascent trajectory was varied such that burnout of the S-IVB established an 80 n.mi. parking orbit. After docking the CSM is used to provide an 80 by 200 n.mi. Hohmann transfer orbit and utilized a second time at apogee to accelerate the spacecraft to circular velocity. The total weight achievable in final orbit by this mode is 33,185 pounds. The weight required for a 45-day mission and deboost is the same as the previous case or 27,590 pounds. The difference between these two

TABLE 5-4. CREW/MISSION EXPENDABLES (TO 45 DAYS)

ITEM	RATE	WEIGHT INCREASE FOR 45-DAY MISSION * (3-MAN CREW) (LBS)
LIFE SUPPORT EXPENDABLES	12.1 LBS/MAN DAY	1,125
	5 LBS/MAN DAY	465
LIFE SUPPORT DRY WEIGHT		
FUEL CELLS REACTANTS	40 LBS/MISSION DAY	1,240
FUEL CELLS TANKAGE	20 LBS/MISSION DAY	620
REACTION CONTROL	60 LBS/MISSION DAY	1,860
	TOTAL	5,310 LBS

*CSM WEIGHTS INCLUDE EXPENDABLES FOR 14 DAY MISSION AND 3 MAN CREW.
THIS COLUMN THUS SHOWS WEIGHTS EQUIVALENT TO 31 MISSION DAYS.

TABLE 5-5. CSM WEIGHT REQUIRED FOR DEBOOST FROM 200 N.MI. CIRCULAR ORBIT

ITEM	WEIGHT
Command Module	11,000 lb
Service Module	10,200
Propellant for 100 fps midcourse maneuver	210
Propellant for 395 fps* deboost impulse	870
Total	22,280 lb

*Deboost assumed 359 fps to be taken from circular velocity to reduce perigee altitude to the earth's surface and then increased by 10 percent for performance contingencies.

weights, then, represents the discretionary payload of 5,595 pounds. A typical sequence of events is given for this flight profile in Table 5-6. Table 5-7 presents the time history of a descent profile from this orbit for impact in the Samoan impact area.

By a similar technique but with an initial flight azimuth of 44 degrees from true north an orbital inclination of 50.3 degrees is realized. The discretionary payload is then reduced to 3340 pounds. The mission profile is given in Table 5-8 and the descent profile in Table 5-9.

The performance capability of an unmanned Saturn IB was determined for a direct ascent flight to a 200 n.mi. circular orbit assuming a due east launch. It was further assumed that in lieu of the CSM, the payload would be covered with a 530 pound shroud. The payload shroud was injected into orbit (as a reliability measure and since relatively little payload would otherwise be gained). The total weight on orbit, including the S-IVB was 62,238 pounds. For an S-IVB stage jettison weight of 35,515 pounds, which allows the payload to be placed inside the LEM adapter and covered by 530-pound shroud, the net payload is 26,723 pounds. The flight profile for this mission is given in Table 5-10.

5.1.1.2 Saturn V Launch Vehicle

The direct ascent burnout weight capability of the Saturn V launch vehicle is 273,495 pounds for a due east launch. This exceeds the current 110,000 pounds stack weight limit of the S-IVB in the Apollo application. The S-IVB jettison weight is 40,539 pounds including the LEM adapter and flight performance reserve. A breakdown of this weight is given in Table 3-3. The CSM weight required for the 45-day mission and for deboost will be the same as for the Saturn IB, i.e., 27,590 pounds. The net payload is therefore 205,366 pounds if the S-IVB stage stack limitation is disregarded. The flight profile is given in Table 5-11.

[redacted]

TABLE 5-6. SATURN IB REFERENCE TRAJECTORIES; 200 N MI CIRCULAR ORBIT BY HOHMANN TRANSFER (PERIGEE = 80 N MI, i = 2[illegible].5 DEG)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1] (DEG)	AZIMUTH[2] (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	—	1,298,819
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	10.00	28.65	-80.64	1,345	506	4.50	90.0	1,234,545
3.	SHUTDOWN OF S-IB INBOARD ENGINES	134.72	28.64	-80.11	7,242	176,584	27.36	90.5	434,345
4.	SHUTDOWN OF S-IB OUTBOARD ENGINES (BURNOUT) BEGIN COAST	141.64	28.64	-80.00	7,623	199,816	26.34	90.6	412,472
5.	JETTISON OF S-IB; S-IVB IGNITION; MIXTURE RATIO = 5.0; CONSTANT ATTITUDE RATE	145.94	28.64	-79.93	7,564	214,082	25.50	90.6	306.646[3]
6.	JETTISON ULLAGE CASES AND THERMOLAG, CHANGE MIXTURE RATIO TO 5.5	155.94	28.64	-79.75	7,648	245,726	23.69	90.8	301,599[4]
7.	JETTISON LAUNCH ESCAPE SYSTEM	165.94	28.64	-79.58	7,772	275,618	21.96	90.9	287,963[5]
8.	CHANGE MIXTURE RATIO TO 4.7	440.00	28.22	-71.48	16,341	557,566	-1.18	95.7	138,981
9.	S-IVB BURNOUT (80 N MI PARKING ORBIT) END INTEGRATED TRAJECTORY	595.54	27.11	-62.61	25,665	486.007	0.00	100.2	69,755
10.	SERVICE MODULE IGNITION FOR PERIGEE MANEUVER	1,795.54[10]	-5.12	11.24	25,665	486,007	0.00	118.1[1]	34,770[6]
11.	SERVICE MODULE BURNOUT, COAST TO APOGEE (7)	1,807.05	-5.12	11.24	25,879	486,007	0.00	118.1	33,968
12.	SERVICE MODULE IGNITION FOR APOGEE MANEUVER	4,495.05	5.12	0.04	25,016	1,215,220	0.00	61.9	33,968
13.	SERVICE MODULE BURNOUT	4,506.29	5.12	0.04	25,229	1,215,220	0.00	61.9	33,185
14.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER[11]	0.00	28.66	35.00	25,229	1,215,220	0.00	90.0	22,280[8]
15.	SERVICE MODULE BURNOUT	12.49	28.66	35.00	24,870	1,215,220	0.00	90.0	21,410
16.	COMMAND MODULE RE-ENTRY	1,607.20	-7.87	132.79	25,831	400,000	-1.52	117.5	11,000[9]

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) AFTER JETTISON EVENT OF 105,826 LB
4) AFTER JETTISON EVENT OF 235 LB
5) AFTER JETTISON EVENT OF 8,200 LB
6) AFTER JETTISON EVENT OF 34,985 LB
7) EACH S/M OPERATION PROVIDES AN ADDITIONAL 10 PERCENT VELOCITY CONTINGENCY OVER THE VELOCITY INCREMENT REQUIRED.
8) THE DIFFERENCE BETWEEN THE WEIGHTS OF EVENTS 13 AND 14 REPRESENTS PAYLOAD LEFT ON ORBIT OR EXPENDED DURING STAY.
9) AFTER JETTISON EVENT OF 10,410 LB WHICH REPRESENTS THE SPENT S/M PLUS 210 LB (100 FPS) FOR MIDCOURSE CORRECTIONS.
10) ALLOWS 20-MIN COAST FOR S/M DOCKING MANEUVER.
11) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-7.

[redacted]

TABLE 5-7. DESCENT FROM 200 N MI ORBIT; i = 28.5 DEGREES

TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1] (DEG)	AZIMUTH[1] (DEG)
0	28.66	35.00	24,870	1,215,220	0.00	90.0
200	27.88	48.76	24,889	1,198,212	-0.37	96.9
400	25.58	62.15	24,946	1,148,090	-0.72	103.3
600	21.94	74.96	25,037	1,067,526	-1.04	108.8
800	17.20	87.10	25,160	960,856	-1.30	113.2
1000	11.60	98.68	25,307	833,889	-1.49	116.3
1200	5.42	109.89	25,471	693,632	-1.61	118.1
1400	-1.09	121.02	25,644	547,928	-1.63	118.5
1600	-7.63	132.37	25,815	405,018	-1.56	117.6
1607.2	-7.87	132.79	25,821	400,000	-1.55	117.5

1) INERTIAL QUANTITIES

TABLE 5-8. SATURN IB REFERENCE TRAJECTORIES; 200 N MI CIRCULAR ORBIT BY HOHMANN TRANSFER
(PERIGEE = 80 N MI, i = 50.3 DEG)

[redacted]

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY 1) (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE 1) (DEG)	AZIMUTH 2) (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	—	1,296,454
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	10.00	28.65	-80.64	1,345	510	4.53	44.0	1,232,181
3.	SHUTDOWN OF S-IB INBOARD ENGINES	134.72	28.98	-80.27	6,980	177,327	28.48	44.0	431,980
4.	SHUTDOWN OF S-IB OUTBOARD ENGINES (BURNOUT) BEGIN COAST	141.64	29.06	-80.19	7,357	200,629	27.36	44.0	410,107
5.	JETTISON OF S-IB; S-IVB IGNITION; MIXTURE RATIO = 5.0; CONSTANT ATTITUDE RATE	145.94	29.10	-80.14	7,295	214,935	26.49	44.1	300,281 3)
6.	JETTISON ULLAGE CASES AND THERMO LAG, CHANGE MIXTURE RATIO TO 5.5	155.94	29.21	-80.02	7,375	246,662	24.59	44.1	299,234 4)
7.	JETTISON LAUNCH ESCAPE SYSTEM	165.94	29.33	-79.89	7,495	276,624	22.76	44.2	285,598 5)
8.	CHANGE MIXTURE RATIO TO 4.7	440.00	34.32	-73.75	16,126	556,384	-1.37	48.5	136,617
9.	BURNOUT S-IVB (80 N MI PARKING ORBIT) END INTEGRATED TRAJECTORY	595.54	39.37	-65.93	25,665	486,057	0.00	53.4	67,391
10.	SERVICE MODULE IGNITION FOR PERIGEE MANEUVER	1,795.54 10)	31.32	36.20	25,665	486,057	0.00	131.7 1)	32,406 6)
11.	SERVICE MODULE BURNOUT, COAST TO APOGEE 7)	1,307.05	31.32	36.20	25,879	486,057	0.00	131.7	31,659
12.	SERVICE MODULE IGNITION FOR APOGEE MANEUVER	4,495.05	-31.32	155.00	25,016	1,215,220	0.00	48.3	31,659
13.	SERVICE MODULE BURNOUT	4,506.29	-31.32	155.00	25,229	1,215,220	0.00	48.3	30,929
14.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER 11)	0.00	0.00	31.00	25,229	1,215,220	0.00	140.3	22,280 8)
15.	SERVICE MODULE BURNOUT	12.49	0.00	31.00	24,870	1,215,220	0.00	140.3	21,410
16.	COMMAND MODULE RE-ENTRY	1,659.40	-46.46	143.62	25,831	400,000	-1.52	67.6	11,000 9)

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) AFTER JETTISON EVENT OF 105,826 LB
4) AFTER JETTISON EVENT OF 235 LB
5) AFTER JETTISON EVENT OF 8,200 LB
6) AFTER JETTISON EVENT OF 34,985 LB
7) EACH S/M OPERATION PROVIDES AN ADDITION 10 PERCENT VELOCITY CONTINGENCY OVER THE VELOCITY INCREMENT REQUIRED
8) THE DIFFERENCE BETWEEN THE WEIGHTS OF EVENTS 13 AND 14 REPRESENTS PAYLOAD LEFT ON ORBIT OR EXPENDED DURING STAY
9) AFTER JETTISON EVENT OF 10,410 LB WHICH REPRESENTS THE SPENT S/M PLUS 210 LB (100 FPS) FOR MIDCOURSE CORRECTIONS
10) ALLOWS 20-MIN COAST FOR S/M DOCKING MANEUVER
11) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-9.

[redacted]

TABLE 5-9. DESCENT FROM 200 N.MI. ORBIT; i = 50.3 DEGREES

TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1)] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1)] (DEG)	AZIMUTH[1)] (DEG)
0	0.00	31.00	24,860	1,215,220	0.00	140.3
200	-9.94	38.47	24,878	1,200,496	-0.38	139.6
400	-19.67	46.48	24,933	1,156,774	-0.75	137.3
600	-28.96	55.65	25,021	1,085,437	-1.07	133.2
800	-37.42	66.79	25,139	988,907	-1.34	126.6
1000	-44.43	80.82	25,283	870,783	-1.53	116.8
1200	-49.09	98.31	25,445	735,988	-1.63	103.4
1400	-50.40	118.36	25,618	590,859	-1.64	87.5
1600	-47.89	138.20	25,793	443,104	-1.56	71.8
1659.4	-46.46	143.62	25,844	400,000	-1.52	67.6

1) INERTIAL QUANTITIES

TABLE 5-10. UNMANNED SATURN IB REFERENCE TRAJECTORY; 200 N.MI. BY DIRECT ASCENT (i = 28.5 DEG)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY 1) (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE 1) (DEG)	AZIMUTH 2) (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	-	1,283,102
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	10.00	28.65	-80.64	1,345	533	4.72	90.0	1,218,828
3.	SHUTDOWN OF S-IB INBOARD ENGINES	134.72	28.65	-80.22	7,019	209,901	39.29	90.4	418,628
4.	SHUTDOWN OF S-IB OUTBOARD ENGINES (BURNOUT), BEGIN COAST	141.64	28.64	-80.13	7,378	241,220	38.59	90.4	307,855
5.	JETTISON OF S-IB; IGNITION; MIXTURE RATIO = 5.0; CONSTANT ATTITUDE RATE	145.94	28.64	-80.08	7,295	260,726	37.83	90.5	290,929 3)
6.	JETTISON OF ULLAGE CASES AND THERMOLAG, CHANGE MIXTURE RATIO TO 5.5	155.94	28.64	-79.94	7,332	304,783	36.27	90.6	285,882 4)
7.	CHANGE MIXTURE RATIO TO 4.7	440.00	28.33	-72.92	15,137	1,109,204	5.50	94.9	131,454
8.	BURNOUT OF S-IVB (200 N.MI. CIRCULAR ORBIT) END INTEGRATED TRAJECTORY	595.54	27.42	-64.74	25,233	1,215,643	0.00	99.2	62,236

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) AFTER JETTISON EVENT OF 105,826 LB
4) AFTER JETTISON EVENT OF 235 LB

[redacted]

TABLE 5-11. SATURN V REFERENCE TRAJECTORIES, LOW-EARTH ORBITS (200 N MI CIRCULAR, i = 28.5 DEG)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY 1) (FT/SEC)	ALTITUDE (FT)	FLIGHT 1) PATH ANGLE (DEG)	AZIMUTH 2) (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	-	6,490,004
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	12.00	28.65	-80.64	1,343	451	3.39	90.0	6,143,543
3.	SHUTDOWN OF S-IC INBOARD ENGINE	154.57	28.64	-80.04	7,709	209,792	26.94	90.6	2,027,385
4.	SHUTDOWN OF S-IC OUTBOARD ENGINES, BEGIN COAST	158.57	28.64	-79.97	8,101	224,988	28.53	90.6	1,934,996
5.	JETTISON OF S-IC, S-II IGNITION, BEGIN PITCH-UP MANEUVER 3), MIXTURE RATIO = 5.0	162.37	28.64	-79.90	8,044	239,479	27.85	90.7	1,553,350 4)
6.	END PITCH-UP MANEUVER, CHANGE MIXTURE RATIO TO 5.4	172,37	28.64	-79.72	8,113	276,251	26.36	90.8	1,529,054
7.	JETTISON S-IC/S-II INTERSTAGE ADAPTER SECTION	188.57	28.64	-79.42	8,270	332,984	24.30	91.0	1,475.844 5)
8.	JETTISON LAUNCH ESCAPE SYSTEM	193.57	28.64	-79.32	8,325	349,854	23.69	91.1	1,454,237 6)
9.	CHANGE MIXTURE RATIO TO 4.7	386.28	28.44	-74.49	12,369	825,033	8.72	94.1	937,476
10.	SHUTDOWN OF S-II, BEGIN COAST	534.19	27.93	-68.73	17,454	1,075,177	5.95	97.2	605,388
11.	JETTISON S-II, S-IVB IGNITION	538.99	27.91	-68.50	17,440	1,083,658	5.71	97.4	504,457 7)
12.	BURNOUT S-IVB, END INTEGRATED TRAJECTORY	1,014.32	22.15	-42.97	25,231	1,214,396	0.00	109.7	273,495 8)
13.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER 11)	0.00	28.66	35.00	25,231	1,215,220	0.00	90.0	22,280 9)
14.	SERVICE MODULE BURNOUT	12.49	28.66	35.00	24,872	1,215,200	0.00	90.0	21,410
15.	COMMAND MODULE RE-ENTRY	1,607.20	-7.87	132.79	25,831	400,000	-1.52	117.5	11,000 10)

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) PITCH-UP OF ONE-DEG/SEC FOR 10-SEC APPROXIMATES OPTIMUM TRAJECTORY, AFTER WHICH A PITCH-DOWN RATE OF 0.0798 DEG/SEC IS MAINTAINED TO S-IVB BURNOUT
4) AFTER JETTISON EVENT OF 381,645 LB
5) AFTER JETTISON EVENT OF 9,770 LB
6) AFTER JETTISON EVENT OF 8,200 LB
7) AFTER JETTISON EVENT OF 100,921 LB
8) S-IVB JETTISON WEIGHT IS 40,539 LB INCLUDING 3800 LB LEM ADAPTER SECTION
9) THE DIFFERENCE BETWEEN 8 AND 9 REPRESENTS THE PAYLOAD LEFT ON ORBIT OR CONSUMED DURING STAY PLUS THE SPENT S-IVB
10) AFTER JETTISON EVENT OF 10,410 LB WHICH REPRESENTS THE SPENT S/M PLUS 210 LB (100 FPS) FOR MIDCOURSE CORRECTIONS
11) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE TABLE 5-7.

[redacted]

5.1.2 Polar Orbits

The data herein is derived for both launch vehicles and for both true polar and 96°.5 inclination (sun synchronous at 200 n. mile altitude) orbits.

Ground traces for each descent trajectory are given in Figure 5-1. Initiation of deboost was timed to provide impact in either of the Apollo impact areas.

5.1.2.1 Saturn IB Launch Vehicle

To establish the maximum payload boundary for the Saturn IB, a trajectory for a polar orbit was computed with a launch azimuth of 182 degrees. Using the CSM to transfer from an 80 n.mi. parking orbit to 200 n.mi. circular orbits results in an injected weight of 25,419 pounds which is less than the 45-day mission requirement of 27,590 pounds. Since this performance was less than that desired for the mission, further performance calculations were abandoned for this Saturn IB configuration. For reference purposes, this profile is given in Table 5-12. Table 5-13 gives a time history of a descent trajectory from a 200 n.mi. polar orbit with impact in the Apollo landing area.

Reference trajectories for the sun-synchronous orbits were derived in a similar manner, i.e., the vehicles were launched at the azimuth which gives the proper inclination at burnout. After transfer to 200 n.mi., the injected weight is 24,568 pounds. Table 5-14 gives the sequence of events for the sun synchronous mission and the descent profile time history is given in Table 5-15.

To effect the required increase in performance, Service Module propulsion was utilized in a suborbital start. This results in a three-stage launch vehicle but necessitates removing the payload from the LEM lab contained within the LEM adapter section. The payload was considered to be contained within the Service

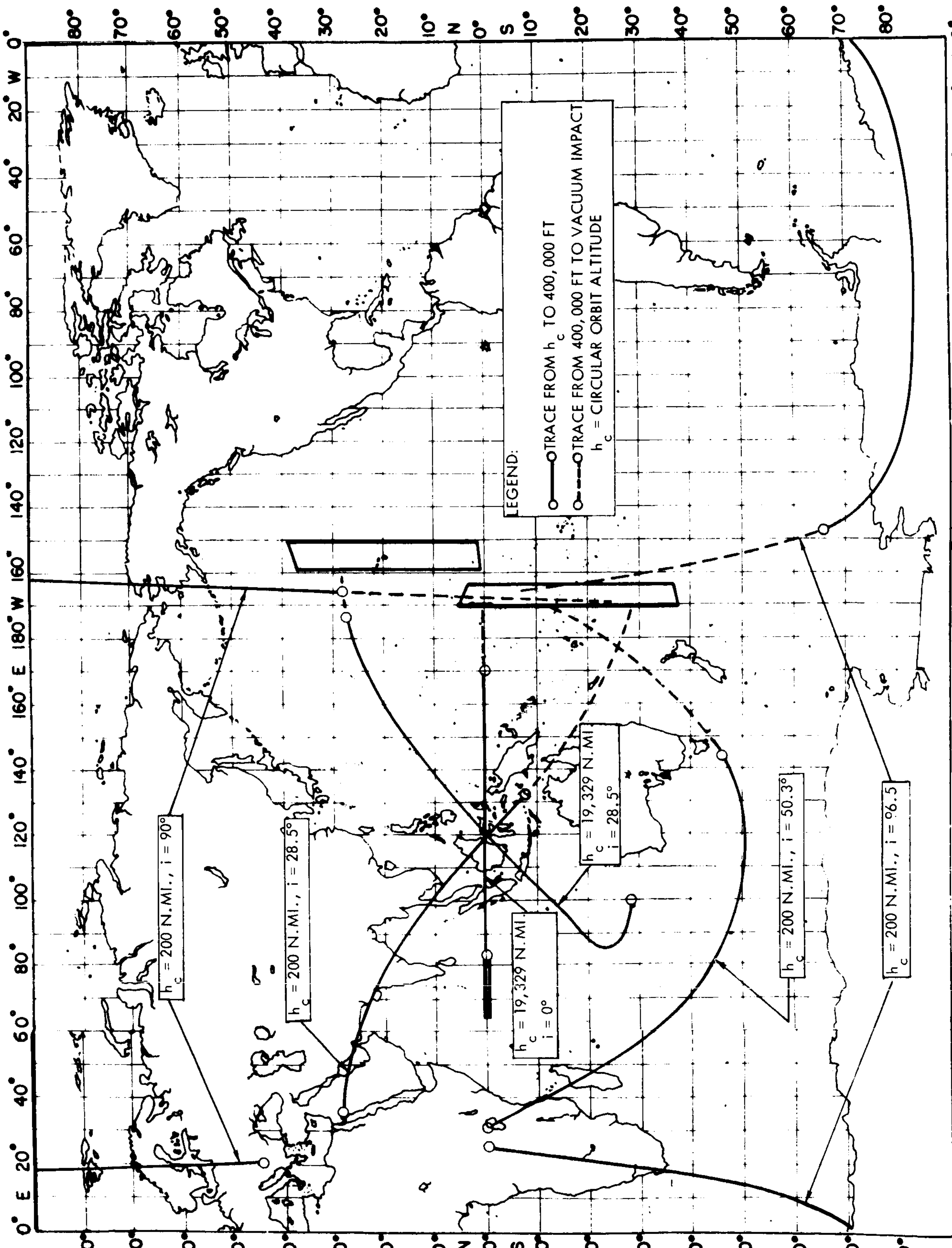


FIGURE 5-1 DEBOOST TRAJECTORY PHASES, AES FLIGHT PROFILES

CONFIDENTIAL

TABLE 5-12. SATURN IB REFERENCE TRAJECTORIES; 200 N MI CIRCULAR ORBIT BY HOHMANN TRANSFER (PERIGEE = 80 N MI, i = 90 DEG)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1] (DEG)	AZIMUTH[2] (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	34.78*	-120.60	1,255	201	0.00	—	1,290,682
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	10.00	34.78	-120.60	1,259	720	4.92	182.0	1,226,409
3.	SHUTDOWN OF S-IB INBOARD ENGINES	134.72	34.31	-120.63	6,259	180,761	33.22	183 0	426,208
4.	SHUTDOWN OF S-IB OUTBOARD ENGINES (BURNOUT), BEGIN COAST	141.64	34.21	-120.64	6,624	204,592	34.21	183.0	404.335
5.	JETTISON OF S-IB; S-IVB IGNITION; MIXTURE RATIO = 5.0; CONSTANT ATTITUDE RATE	145.94	34.15	-120.64	6,554	219,229	30.82	183.1	298,509[3]
6.	JETTISON OF ULLAGE CASES AND THERMOLAG, CHANGE MIXTURE RATIO TO 5.5	155.94	34.00	-120.65	6,617	251,705	28.65	183.1	293,462[4]
7.	JETTISON LAUNCH ESCAPE SYSTEM	165.94	33.84	-120.66	6,722	282,402	26.54	183.2	279,826[5]
8.	CHANGE MIXTURE RATIO TO 4.7	440.00	26.49	-121.19	14,486	569,984	-1.31	183.7	130,844
9.	BURNOUT S-IVB (80 N MI PARKING ORBIT), END INTEGRATED TRAJECTORY	595.54	18.16	-121.76	25,661	486,582	0.00	183.5	61,618
10.	SERVICE MODULE IGNITION FOR PERIGEE MANEUVER	1,795.54[10]	-64.49	-126.77	25,661	486,582	0.00	180.0[1]	26,633[6]
11.	SERVICE MODULE BURNOUT, COAST TO APOGEE 7)	1,807.05	-64.49	-126.77	25,872	486,582	0.00	180.0	26,019
12.	SERVICE MODULE IGNITION FOR APOGEE MANEUVER	4,495.05	64.49	42.03	25,016	1,215,220	0.00	0.0	26,019
13.	SERVICE MODULE BURNOUT	4,506.29	64.49	42.03	25,229	1,215,220	0.00	0.0	25,419
14.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER 11)	0.00	44.86	20.70	25,229	1,215,220	0.00	0.0	22,280[8]
15.	SERVICE MODULE BURNOUT	12.49	44.86	20.70	24,870	1,215,220	0.00	0.0	21,410
16.	COMMAND MODULE RE-ENTRY	1,608.90	28.96	-166.02	25,831	400,000	-1.52	180.0	11,000[9]

* FLIGHT WAS ASSUMED TO BE FROM WESTERN TEST RANGE

1) INERTIAL QUANTITIES

2) RELATIVE QUANTITIES

3) AFTER JETTISON EVENT OF 105,826 LB

4) AFTER JETTISON EVENT OF 235 LB

5) AFTER JETTISON EVENT OF 8,200 LB

6) AFTER JETTISON EVENT OF 34,985 LB

7) EACH S/M OPERATION PROVIDES AN ADDITIONAL 10 PERCENT VELOCITY CONTINGENCY OVER THE VELOCITY INCREMENT REQUIRED.

8) THE DIFFERENCE BETWEEN THE WEIGHTS OF EVENTS 13 AND 14 REPRESENTS PAYLOAD LEFT ON ORBIT OR EXPENDED DURING STAY.

9) AFTER JETTISON EVENT OF 10,410 LB WHICH REPRESENTS THE SPENT S/M PLUS 210 LB (100 FPS) FOR MIDCOURSE CORRECTIONS.

10) ALLOWS 20-MIN COAST FOR S/M DOCKING MANEUVER

11) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-13.

CONFIDENTIAL

CONFIDENTIAL

TABLE 5-13. DESCENT FROM 200 N MI ORBIT; i = 90.0 DEGREES

TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1)] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1)] (DEG)	AZIMUTH[1)] (DEG)
0	44.86	20.70	24,860	1,215,220	0.00	0.0
200	57.73	19.87	24,871	1,214,074	-0.38	0.0
400	70.60	19.03	24,920	1,178,133	-0.73	0.0
600	83.52	18.20	25,007	1,107,550	-1.04	0.0
700	89.98	-162.22	25,062	1,059,945	-1.18	180.0
800	83.46	-162.64	25,126	1,004,807	-1.30	180.0
1000	70.30	-163.47	25,274	874,899	-1.48	180.0
1200	56.95	-164.31	25,442	725,288	-1.59	180.0
1400	43.38	-165.14	25,622	565,588	-1.61	180.0
1600	29.58	-165.98	25,801	406,897	-1.55	180.0
1608.9	28.96	-166.02	25,810	400,000	-1.54	180.0

1) INERTIAL QUANTITIES

[redacted]

CONFIDE[redacted]

TABLE 5-14. SATURN IB REFERENCE T[redacted] N MI CIRCULAR ORBIT BY HOHMANN TRANSFER (PERIGEE = 80 N MI, i = 96.5 DEG)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY 1) (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE 1) (DEG)	AZIMUTH 2) (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	34.78*	-120.60	1,255	201	0.00	—	1,289,790
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	10.00	34.78	-120.60	1,259	721	4.94	189.4	1,225,517
3.	SHUTDOWN OF S-IB INBOARD ENGINES	134.72	34.32	-120.70	6,136	181,182	34.08	190.4	425,316
4.	SHUTDOWN OF S-IB OUTBOARD ENGINES (BURNOUT), BEGIN COAST	141.64	34.22	-120.72	6,499	205,074	32.54	190.4	403,443
5.	JETTISON OF S-IB; S-IVB IGNITION; MIXTURE RATIO = 5.0; CONSTANT ATTITUDE RATE	145.94	34.16	-120.74	6,428	219,748	31.58	190.4	297,617 3)
6.	JETTISON OF ULLAGE CASES AND THERMOLAG, CHANGE MIXTURE RATIO TO 5.5	155.84	34.01	-120.77	6,488	252,309	29.36	190.5	292,570 4)
7.	JETTISON LAUNCH ESCAPE SYSTEM	165.94	33.85	-120.81	6,590	283,085	27.16	190.5	278,934 5)
8.	CHANGE MIXTURE RATIO TO 4.7	440.00	20.59	-122.38	15,382	570,998	-1.35	190.5	129,953
9.	BURNOUT S-IVB (80 N MI PARKING ORBIT), END INTEGRATED TRAJECTORY	595.54	18.34	-123.98	25,662	486,557	0.00	189.9	60,727
10.	SERVICE MODULE IGNITION FOR PERIGEE MANEUVER	1,795.54 10)	-63.50	-143.83	25,662	486,557	0.00	194.2 1)	25,742 6)
11.	SERVICE MODULE BURNOUT, COAST TO APOGEE 7)	1,807.05	-63.50	-143.83	25,876	486,557	0.00	194.2	25,148
12.	SERVICE MODULE IGNITION FOR APOGEE MANEUVER	4,495.05	63.50	24.97	25,016	1,215,220	0.00	14.2	25,148
13.	SERVICE MODULE BURNOUT	4,506.29	63.50	24.97	25,229	1,215,220	0.00	14.2	24,568
14.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER 11)	0.00	0.00	25.00	25,229	1,215,220	0.00	186.5	22,280 8)
15.	SERVICE MODULE BURNOUT	12.49	0.00	25.00	24,870	1,215,220	0.00	186.5	21,410
16.	COMMAND MODULE RE-ENTRY	1,704.10	-66.29	-147.14	25,831	400,000	-1.52	343.7	11,000 9)

* FLIGHT WAS ASSUMED TO BE FROM WESTERN TEST RANGE

1) INERTIAL QUANTITIES

2) RELATIVE QUANTITIES

3) AFTER JETTISON EVENT OF 105,826 LB

4) AFTER JETTISON EVENT OF 235 LB

5) AFTER JETTISON EVENT OF 8,200 LB

6) AFTER JETTISON EVENT OF 34,985 LB

7) EACH S/M OPERATION PROVIDES AN ADDITION 10 PERCENT VELOCITY CONTINGENCY OVER THE VELOCITY INCREMENT REQUIRED

8) THE DIFFERENCE BETWEEN THE WEIGHTS OF EVENTS 13 AND 14 REPRESENTS PAYLOAD LEFT ON ORBIT OR EXPENDED DURING STAY

9) AFTER JETTISON EVENT OF 10,410 LB WHICH REPRESENTS THE SPENT S/M PLUS 210 LB (100 FPS) FOR MIDCOURSE CORRECTIONS

10) ALLOWS 20-MIN COAST FOR S/M DOCKING MANEUVER

11) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-15.

C[redacted]

TABLE 5-15. DESCENT FROM 200 N.MI. ORBIT; i = 96.5 DEGREES

TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1)] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1)] (DEG)	AZIMUTH[1)] (DEG)
0	0.00	25.00	24,860	1,215,220	0.00	186.5
200	-12.87	22.68	24,878	1,201,881	-0.38	186.7
400	-25.75	20.20	24,930	1,162,080	-0.75	187.2
600	-38.66	17.30	25,015	1,096,540	-1.07	188.3
800	-51.57	13.45	25,129	1,006,700	-1.33	190.5
1000	-64.44	7.12	25,269	895,024	-1.52	195.2
1100	-70.78	1.44	25,346	832,136	-1.58	200.0
1200	-76.86	-9.04	25,427	765,352	-1.62	209.7
1300	-81.98	-34.00	25,512	695,416	-1.63	233.8
1400	-83.23	-88.12	25,597	623,186	-1.62	287.2
1500	-79.08	-125.25	25,684	549,633	-1.59	323.5
1600	-73.08	-139.79	25,770	475,836	-1.53	337.2
1704.1	-66.29	-147.14	25,858	400,000	-1.46	343.7

1) INERTIAL QUANTITIES

Module and to be deboosted from orbit with the CSM. The Service Module propellant was accordingly increased to provide the deboost velocity for the added weight.

Launch azimuths of 146 degrees and 182 degrees were assumed, the latter to again establish an outer bound on payload capability. The azimuth of 146 degrees is the most southerly that is probable from ETR (Section 5.1.2.2) and therefore must be combined with a yaw program to attain the proper velocity vector. The preliminary range safety criteria were satisfied in the sense that the nominal stage impact points are in the open ocean area.

Propellant in the S-IVB stage was offloaded and the Service Module loaded to capacity. It was found that offloading the S-IVB by about 40,890 pounds maximized payload. However, a complete propellant distribution study among all three stages was not conducted. It is felt that by doing so some improvement in performance might be realized. Also, payload might be increased through a detailed trajectory shaping optimization.

The sequence of trajectory events is summarized in Table 5-16 for the Saturn IB (three-stage arrangement) launched at the 146 degree azimuth. A coast period of 5.4 seconds was allowed between S-IVB and Service Module operation. The yaw program which provides the polar inclination is given in the footnotes of Table 5-16.

5.1.2.2 Saturn V Launch Vehicle

The payload capabilities of the Saturn V launch vehicle have been computed for 200 n.mi. circular orbits for the various orbital inclinations associated with the ETR launch vacilities. With the Range Safety model given in Section 4.0, inclinations between 28.5 and 50.3 degrees are obtainable by selection of the proper initial flight azimuths. Inclinations greater than 50.3 degrees require a

~~CONFIDENTIAL~~

TABLE 5-16. THREE-STAGE SATURN IB REFERENCE TRAJECTORIES; 200 N.MI. CIRCULAR ORBIT BY HOHMANN TRANSFER (PERIGEE = 80 N.MI., i = 90)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1] (FT/SEC)	ALTITUDE (DEG)	FLIGHT PATH ANGLE[1] (DEG)	AZIMUTH[2] (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	-	1,286,617
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	10.00	28.65	-80.64	1,345	527	4.67	146.0	1,222,344
3.	SHUTDOWN OF S-IB INBOARD ENGINES	134.72	28.31	-80.41	6,534	200,734	38.57	155.7	422,143
4.	SHUTDOWN OF S-IB OUTBOARD ENGINES (BURNOUT) BEGIN COAST	141.64	28.23	-80.37	6,842	229,181	37.52	157.6	400,270
5.	JETTISON OF S-IB; S-IVB IGNITION; MIXTURE RATIO = 5.0; CONSTANT ATTITUDE RATE[3]	145.94	28.18	-80.34	6,761	246,761	36.67	157.6	294,444[4]
6.	JETTISON ULLAGE CASES AND THERMOLAG, CHANGE MIXTURE RATIO TO 5.5	155.94	28.06	-80.29	6,771	286,153	34.79	158.7	289,397[5]
7.	JETTISON LAUNCH ESCAPE SYSTEM	165.94	27.93	-80.24	6,797	323,811	32.89	160.2	275,761[6]
8.	CHANGE MIXTURE RATIO TO 4.7	440.00	22.14	-79.75	14,076	826,188	3.59	182.6	126,780
9.	S-IVB BURNOUT, BEGIN COAST[7]	503.54	18.91	-79.91	17,204	880,373	3.42	184.1	98,445
10.	SERVICE MODULE IGNITION	508.94	18.66	-79.93	17,195	885,496	3.13	184.2	63,460[8]
11.	SERVICE MODULE BURNOUT (80 x 200 N.MI. TRANSFER ORBIT) END INTEGRATED TRAJECTORY	1,061.85	-12.08	-82.26	25,865	485,652	0.00	0.0[1]	24,950
12.	SERVICE MODULE IGNITION FOR APOGEE MANEUVER	3,754.95	12.08	90.26	25,016	1,215,220	0.00	0.0	24,950
13.	SERVICE MODULE BURNOUT	3,763.21	12.08	90.26	25,229	1,215,220	0.00	0.0	24,375
14.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER[9]	0.00	44.86	20.70	25,229	1,215,220	0.00	0.0	24,375[10]
15.	SERVICE MODULE BURNOUT	13.45	44.86	20.70	24,860	1,215,220	0.00	0.0	23,438
16.	COMMAND MODULE REENTRY	1,608.90	28.96	-166.02	25,810	400,000	-1.54	180.0	11,000[11]

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) CONSTANT PITCH-DOWN RATE OF 0.0437 DEG/SEC IS MAINTAINED TO FIRST SM BURNOUT. A YAW PROGRAM OF ONE DEG/SEC BETWEEN 110-134 AND BETWEEN 146 AND 164.1 SEC PROVIDES AN ORBIT INCLINATION OF 90 DEGREES.
4) AFTER JETTISON EVENT OF 105,826 LB
5) AFTER JETTISON EVENT OF 235 LB
6) AFTER JETTISON EVENT OF 8,200 LB
7) S-IVB WAS OFFLOAD BY 40,892 LB
8) AFTER JETTISON EVENT OF 34,985 LB
9) THE TIME HISTORY OF A DESCENT PROFILE IS GIVEN IN TABLE 5-10.
10) ASSUMES THE PAYLOAD WILL BE DEBOOSTED FROM ORBIT.
11) AFTER JETTISON EVENT OF 12,438 LB OF WHICH 10,410 LB REPRESENT SPENT SM; THUS, 2028 LB IS ALL THAT CAN BE ALLOTTED TO CREW AND PAYLOAD.

~~CONFIDENTIAL~~

yaw program during the boost phases. This section presents the results of the study involving the most practical trajectories by which orbits of high inclination can be obtained.

A steering program during the boost phases is necessary to yaw the velocity vector to the desired direction. When a yaw program is to be incorporated, two objectives are sought in order to enhance the payload capability. First is to minimize the angle through which the velocity vector is turned and secondly, to achieve this change while the velocity magnitude is small. Range Safety factors determine the minimum turn angle by restricting launch azimuth and usually a combination of Range Safety factors and the booster's structural integrity delays the initiation of the yaw maneuver.

Orbits with initial flight azimuths of 44, 114, and 146 degrees were investigated. Each corridor was investigated to the extent required to determine its potentiality or practicality. For the azimuth providing the most promising trajectory, dispersion ellipses of the stage impact points were determined along with the performance penalties associated with placing them off land masses in conformity to the Range Safety Model. The trace of instantaneous impact points (IIP) of a Saturn V trajectory when launched at an azimuth of 44 degrees skirts the eastern coast of the U. S. and is midway thru S-II operation before a yaw program can be initiated to turn into a polar inclination. This forces the IIP trace around Newfoundland and Labrador in compliance with the Range Safety Model. The payload decrement associated with the yaw maneuver is large. In the case cited, the total weight in a 200 n.mi. polar orbit is 90,480 pounds. The yaw program of 1 deg/sec (with reference to vehicle attitude) is from 392 seconds from liftoff to 480.3 seconds. Without a yaw program, the corresponding weight is

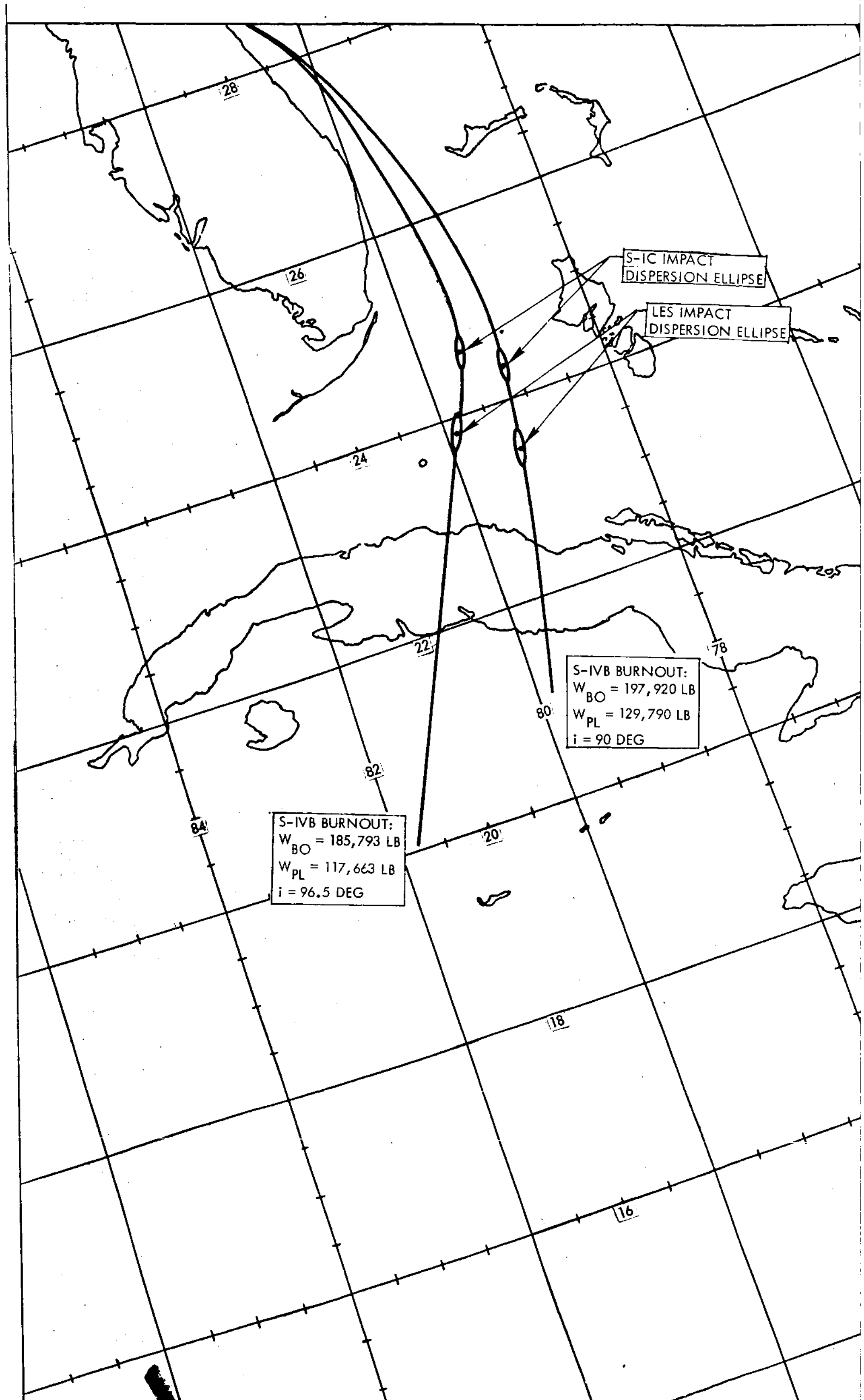
263,190 pounds and the resulting inclination is 50.3 degrees. Certain increases in payload could be expected from a more detailed study but greater gains were felt available at other azimuths.

The orbit inclinations resulting from a launch azimuth within the azimuth sector (44° to 114°) will vary from about 28.5° to 50.3°. More southerly azimuths than 114° are considered impractical because of jettisoned masses impacting on land with the exception of a corridor along an azimuth of 146 degrees. Trajectories along both these paths were investigated.

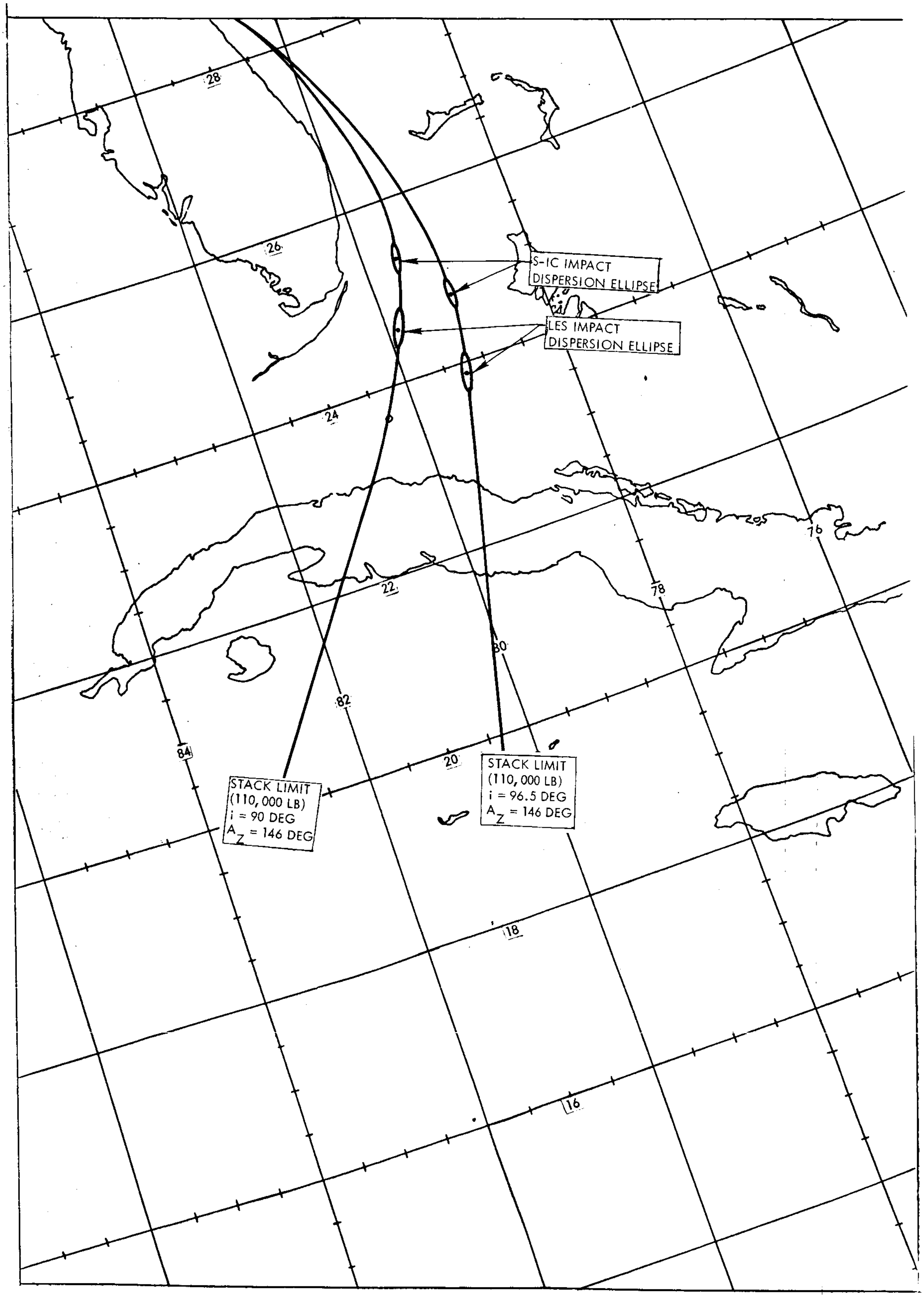
For an azimuth of 114 degrees yaw initiation is delayed until after S-II ignition, or perhaps a few seconds longer. The trace of IIP gradually turns south. The impact of the S-II stage is well into South America, making the trajectory unacceptable. By removing the S-IVB stage (and relocating the IU) and using the same flight sequence an acceptable trajectory is realized since the S-II stage is injected into orbit. The total weight into a polar orbit is then 187,208 pounds of which 100,580 pounds is the spent S-II.

Figure 5-2 is a map of Southern Florida and the Caribbean and shows the corridor accessible with a launch azimuth of 146 degrees and a yaw program. The corridor is attractive because it offers the best compromise relative to both factors, discussed earlier, which minimize the performance penalties. The advantages are somewhat lessened due to the necessity of preventing the LES dispersion ellipse from impacting in Cuba. This was accomplished by a 4-second early shutdown of the S-IC stage. The net effect, however, is the trajectory which provides the maximum payload weight in a polar orbit.

FIGURE 5.2. SA



28
26
24
22
20
18
16
84
82
80
78
S-IC IMPACT DISPERSION ELLIPSE
LES IMPACT DISPERSION ELLIPSE
S-IVB BURNOUT:
W_{BO} = 197,920 LB
W_{PL} = 129,790 LB
i = 90 DEG
S-IVB BURNOUT:
W_{BO} = 185,793 LB
W_{PL} = 117,663 LB
i = 96.5 DEG


28
26
24
22
20
18
16
84
82
80
78
76
S-IC IMPACT
DISPERSION ELLIPSE
LES IMPACT
DISPERSION ELLIPSE
STACK LIMIT
(110,000 LB)
i = 90 DEG
A$_Z$ = 146 DEG
STACK LIMIT
(110,000 LB)
i = 96.5 DEG
A$_Z$ = 146 DEG

Traces are also shown in the figure for the polar and sun synchronous trajectories in accordance with the S-IVB stack limit. The dispersion ellipses are illustrated for S-IC and LES inpact.

Figure 5-3 is a map covering the area to the south and includes the impact dispersion ellipses of the spent S-II for the two inclinations. The stack limit trajectories are not shown. Instead, the two stage Saturn V launched at 114 deg and steered polar is shown. Since inpact of the S-IC and LES occur in the normal impact area for boosters, the impact dispersions were not evaluated.

The range increments are shown in Figure 5-4 for the impact point dispersion analysis. Included in the figure are the perturbations which contributed to dispersions. The polar orbit trajectory was simulated (open-loop) and perturbed per these conditions.

The polar orbit mission profile is given in Table 5-17 and the mission profile for the sun synchronous orbit in Table 5-18.

Alternate profiles are offered in Tables 5-19 and 5-20 for the same two missions and correspond to a S-IVB stack weight limit of 110,000 pounds. The lighter upper body weight required the S-IC shutdown to be 8 seconds earlier than nominal to prevent LES impact on Cuba. This does not represent a performance penalty in this case since S-IVB shutdown occurs with many pounds of unconsumed propellant.

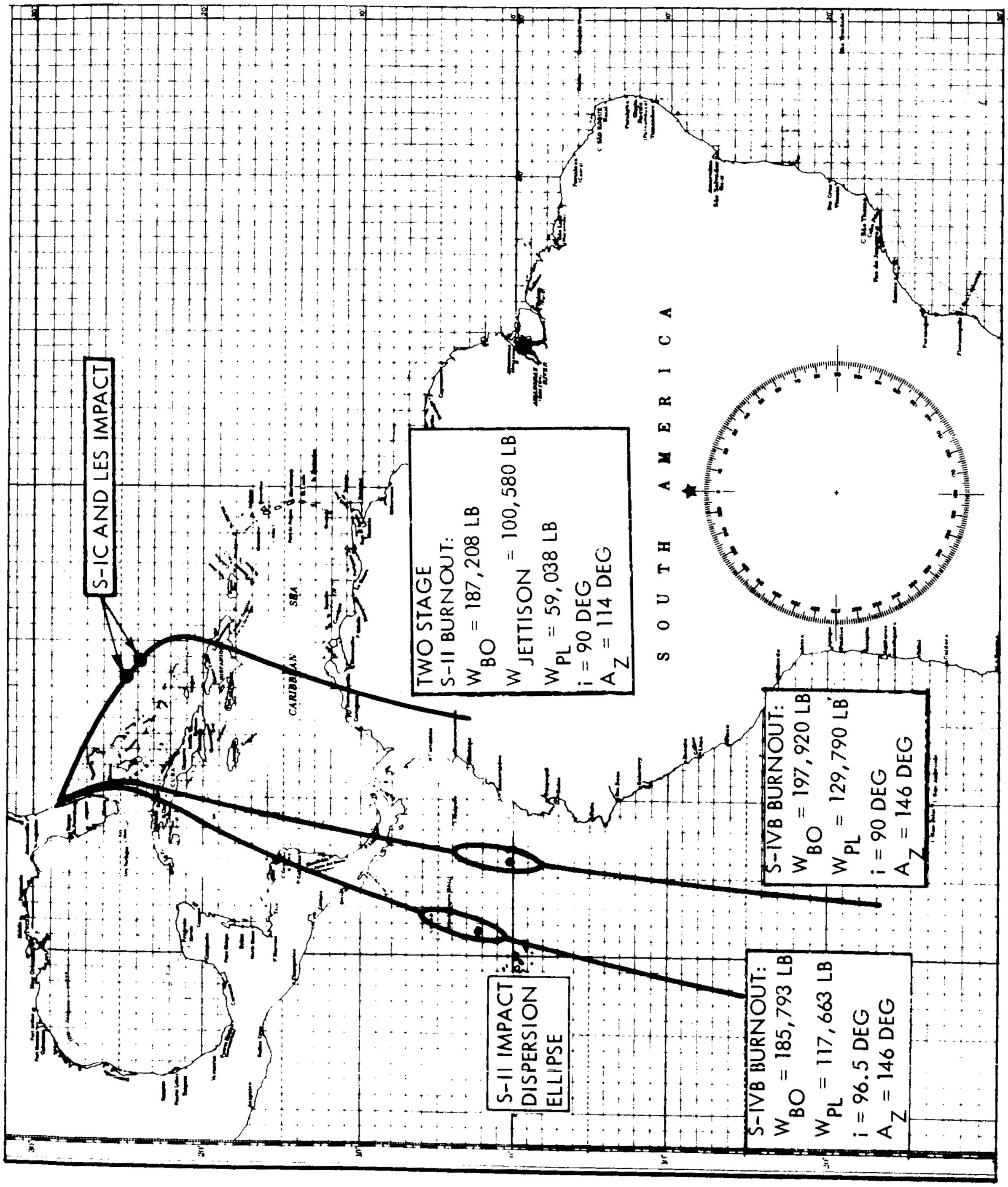
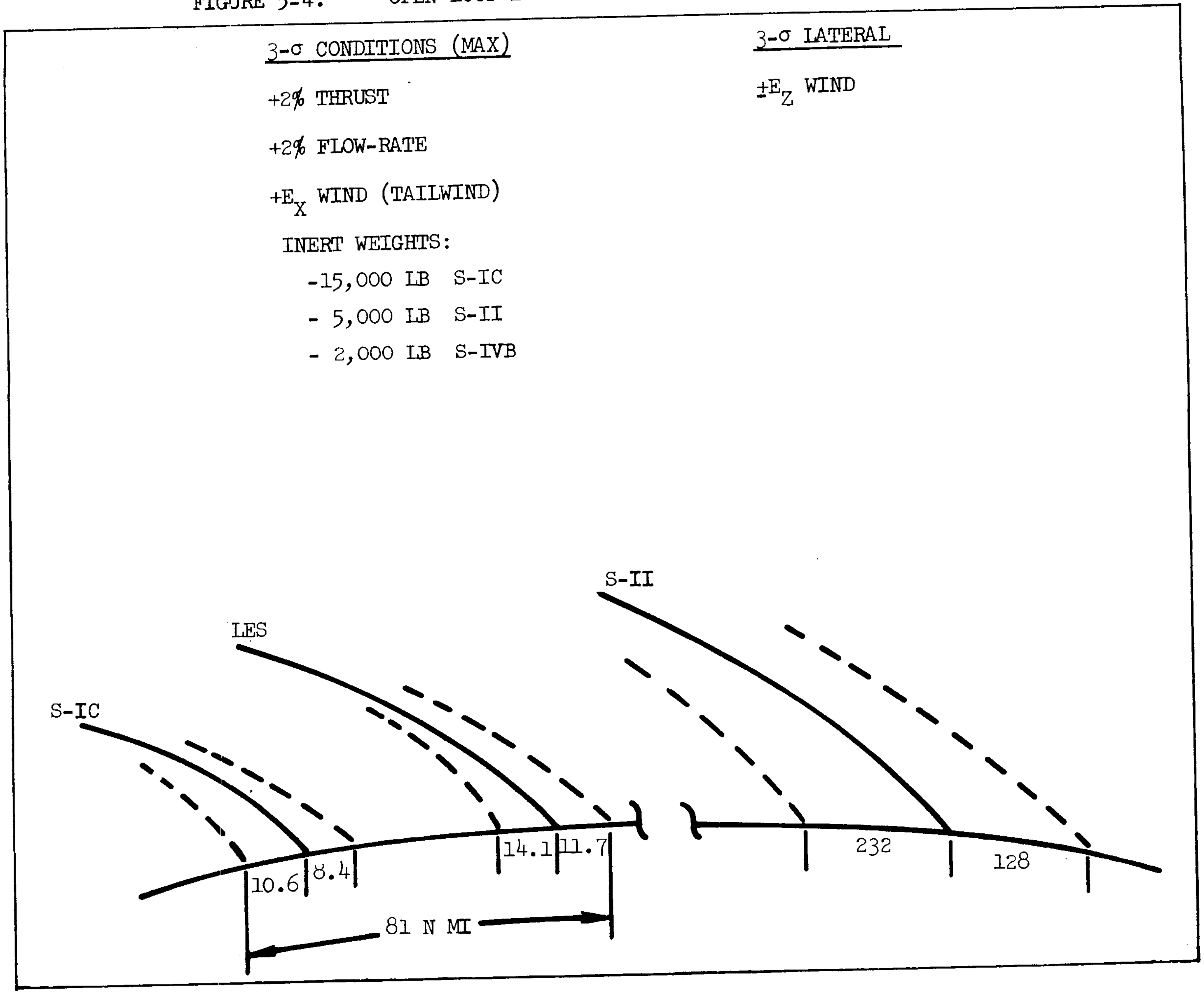


FIGURE 5-3. APOLLO EXTENSION SYSTEMS, POLAR ORBITS

FIGURE 5-4. OPEN-LOOP IMPACT POINT DISPERSION



[redacted]

Table 5-17. SATURN V REFERENCE TRAJECTORIES, LOW-EARTH ORBITS (200 N MI CIRCULAR, i = 90.0 DEG*)

	TRAJECTORY EVENT	(SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	1) VELOCITY (FT/SEC)	ALTITUDE (FT)	FLIGHT 1) PATH ANGLE (DEG)	2) AZIMUTH (DEG)	VEHICLE WEIGHT (LB)	
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	—	6,414,428	
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	12.00	28.65	-80.64	1,343	484	3.63	146.0	6,067,967	
3.	SHUTDOWN OF S-IC INBOARD ENGINE	150.57	28.32	-80.43	6,416	227,090	43.37	160.9	6,067,296	
4.	SHUTDOWN OF S-IC OUTBOARD ENGINES,[3)] BEGIN COAST	154.57	28.28	-80.42	6,712	244,982	42.85	163.0	1,974,906	
5.	JETTISON OF S-IC, S-II IGNITION, BEGIN PITCH-UP MANEUVER,[4)] MIXTURE RATIO = 5.0	156.37	28.23	-80.40	6,631	262,065	42.44	163.1	1,477,774	5)
6.	END PITCH-UP MANEUVER, CHANGE MIXTURE RATIO TO 5.4	168.37	28.12	-80.36	6,610	305,651	40.50	164.4	1,453,478	
7.	JETTISON S-IC/S-II INTERSTAGE ADAPTER SECTION	184.57	27.92	-80.31	6,593	373,132	38.04	167.2	1,400,266	6)
8.	JETTISON LAUNCH ESCAPE SYSTEM	189.57	27.86	-80.29	6,595	393,207	37.26	168.1	1,378,660	7)
9.	CHANGE MIXTURE RATIO TO 4.7	382.28	24.34	-80.31	9,794	947,847	12.25	186.4	861,900	
10.	SHUTDOWN OF S-II, BEGIN COAST	530.19	19.64	-81.03	15,409	1,203,034	6.56	188.8	529,812	
11.	JETTISON S-II, S-IVB IGNITION	534.99	19.45	-81.06	15,394	1,211,130	6.24	188.8	428,891	8)
12.	BURNOUT S-IVB, END INTEGRATED TRAJECTORY	1,010.32	-4.72	-64.03	25,228	1,215,571	0.00	183.7	197,920	9)
13.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER[12)]	0.00	44.36	20.70	25,228	1,215,571	0.00	0.0	22,280	10)
14.	SERVICE MODULE BURNOUT	12.49	44.36	20.70	24,870	1,215,571	0.00	0.0	21,410	
15.	COMMAND MODULE RE-ENTRY	1,638.90	28.96	-166.02	25,831	400,000	-1.52	180.0	11,000	11)

*NO STACK LIMIT CONSTRAINT ON S-IVB

1) INERTIAL QUANTITIES

2) RELATIVE QUANTITIES

3) S-IC WAS SHUTDOWN 4 SEC EARLY TO PLACE SPENT BOOSTER AND LES IN A SAFE IMPACT AREA

4) PITCH-UP OF ONE-DEG/SEC FOR 10-SEC APPROXIMATES OPTIMUM TRAJECTORY, AFTER WHICH A PITCH-DOWN RATE OF 0.0778 DEG/SEC IS MAINTAINED TO S-IVB BURNOUT. A YAW PROGRAM OF ONE-DEG/SEC BETWEEN 120 AND 150 SEC AND BETWEEN 159 AND 172.4 SEC PROVIDES AN ORBIT INCLINATION OF 90 DEG.

5) AFTER JETTISON EVENT OF 497,132 LB

6) AFTER JETTISON EVENT OF 9,770 LB

7) AFTER JETTISON EVENT OF 8,200 LB

8) AFTER JETTISON EVENT OF 100,921 LB

9) S-IVB JETTISON WEIGHT IS 40,539 LB INCLUDING 3800 LB LEM ADAPTER SECTION

10) THE DIFFERENCE BETWEEN 9 AND 10 REPRESENTS THE PAYLOAD LEFT ON ORBIT OR CONSUMED DURING STAY PLUS THE SPENT S-IVB

11) AFTER JETTISON EVENT OF 10,410 LB WHICH REPRESENTS THE SPENT S/M PLUS 210 LB (100 FPS) FOR MIDCOURSE CORRECTIONS

12) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-13

[redacted]

[redacted]

TABLE 5-18. SATURN V REFERENCE TRAJECTORIES, LOW-EARTH ORBITS (200 N MI CIRCULAR, i = 96.5 DEG*)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1] (DEG)	AZIMUTH[2] (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	—	6,398,883
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	12.00	28.65	-80.64	1,343	491	3.69	146.0	6,052,423
3.	SHUTDOWN OF S-IC INBOARD ENGINE	150.57	28.37	-80.47	6,288	239,078	49.71	163.5	2,051,751
4.	SHUTDOWN OF S-IC OUTBOARD ENGINES,[3] BEGIN COAST	154.57	28.33	-80.46	6,569	258,592	49.30	166.1	1,959,362
5.	JETTISON OF S-IC. S-II IGNITION, BEGIN PITCH-UP MANEUVER,[4] MIXTURE RATIO = 5.0	158.37	28.29	-80.45	6,479	277,258	48.65	166.2	1,462,230 [5]
6.	END PITCH-UP MANEUVER, CHANGE MIXTURE RATIO TO 5.4	168.37	28.19	-80.42	6,427	325,038	47.17	167.8	1,437,934
7.	JETTISON S-IC/S-II INTERSTAGE ADAPTER SECTION	184.57	28.02	-80.39	6,338	399,297	44.84	171.7	1,384,723 [6]
8.	JETTISON LAUNCH ESCAPE SYSTEM	189.57	27.97	-80.38	6,314	421,390	44.06	172.9	1,363,116 [7]
9.	CHANGE MIXTURE RATIO TO 4.7	382.26	24.81	-80.93	9,096	1,022,014	13.69	187.2	846,356
10.	SHUTDOWN OF S-II, BEGIN COAST	530.19	20.44	-82.44	14,920	1,273,018	6.31	198.1	514,268
11.	JETTISON S-II, S-IVB IGNITION	534.99	20.26	-82.50	14,905	1,280,533	5.97	198.1	413,347 [8]
12.	BURNOUT S-IVB, END INTEGRATED TRAJECTORY	1,010.32	-3.19	-88.78	25,228	1,214,918	0.00	190.1	182,375 [9]
13.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER[12]	0.00	0.00	25.00	25,228	1,214,918	0.00	186.5	22,280 [10]
14.	SERVICE MODULE BURNOUT	12.49	0.00	25.00	24,870	1,214,918	0.00	186.5	21,410
15.	COMMAND MODULE RE-ENTRY	1,704.1	-66.29	-147.14	25,831	400,000	-1.52	343.7	11,000 [11]

*NO STACK LIMIT CONSTRAINT ON S-IVB

1) INERTIAL QUANTITIES

2) RELATIVE QUANTITIES

3) S-IC WAS SHUTDOWN 4 SEC EARLY TO PLACE SPENT BOOSTER AND LES IN A SAFE IMPACT AREA

4) PITCH-UP OF ONE-DEG/SEC FOR 10-SEC APPROXIMATES OPTIMUM TRAJECTORY, AFTER WHICH A PITCH-DOWN RATE OF 0.0785 DEG/SEC IS MAINTAINED TO S-IVB BURNOUT. A YAW PROGRAM OF ONE-DEG/SEC BETWEEN 120 AND 150 SEC AND BETWEEN 159 AND 181.9 SEC PROVIDES AN ORBIT INCLINATION OF 96.5 DEG

5) AFTER JETTISON EVENT OF 497,132 LB

6) AFTER JETTISON EVENT OF 9,770 LB

7) AFTER JETTISON EVENT OF 8,200 LB

8) AFTER JETTISON EVENT OF 100,921 LB

9) S-IVB JETTISON WEIGHT IS 40,539 LB INCLUDING 3800 LB LEM ADAPTER SECTION

10) THE DIFFERENCE BETWEEN 9 AND 10 REPRESENTS THE PAYLOAD LEFT ON ORBIT OR CONSUMED DURING STAY PLUS THE SPENT S-IVB

11) AFTER JETTISON EVENT OF 10,410 LB WHICH REPRESENTS THE SPENT S/M PLUS 210 LB (100 FPS) FOR MIDCOURSE CORRECTIONS

12) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-15

[redacted]

[redacted]

TABLE 5-19. SATURN V REFERENCE TRAJECTORIES, LOW-EARTH ORBITS (200 N MI CIRCULAR, i = 90 DEG, S-IVB STACK LIMIT)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY 1) (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE 1) (DEG)	AZIMUTH 2) (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	—	6,367,048
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	12.00	28.65	-80.64	1,343	505	3.79	146.0	6,020,588
3.	SHUTDOWN OF S-IC INBOARD ENGINE	146.57	28.35	-80.44	6,194	215,664	43.93	156.2	2,135,403
4.	SHUTDOWN OF S-IC OUTBOARD ENGINES 3), BEGIN COAST	150.57	28.31	-80.42	6,476	233,132	43.49	160.3	2,043,013
5.	JETTISON OF S-IC, S-II IGNITION, BEGIN PITCH-UP MANEUVER 4), MIXTURE RATIO = 5.0	154.37	28.27	-80.41	6,394	249,809	42.76	160.4	1,430,394 5)
6.	END PITCH-UP MANEUVER, CHANGE MIXTURE RATIO TO 5.4	164.37	28.16	-80.37	6,378	292,374	41.14	161.8	1,406,098
7.	JETTISON S-IC/S-II INTERSTAGE ADAPTER SECTION	180.57	27.98	-80.31	6,358	358,446	38.72	164.9	1,352,888 6)
8.	JETTISON LAUNCH ESCAPE SYSTEM	185.57	27.92	-80.29	6,358	378,106	37.95	166.0	1,331,261 7)
9.	CHANGE MIXTURE RATIO TO 4.7	378.28	24.51	-80.30	9,749	915,998	11.66	186.9	814,520
10.	SHUTDOWN OF S-II, BEGIN COAST	526.19	19.71	-81.07	15,985	1,151,213	5.76	188.9	482,432
11.	JETTISON S-II, S-IVB IGNITION	530.99	19.51	-81.10	15,971	1,158,559	5.46	186.9	361,511 8)
12.	BURNOUT S-IVB, END INTEGRATED TRAJECTORY	933.77	-1.38	-83.65	25,228	1,215,383	0.00	183.7	185,793 9)
13.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER 10)	0.00	44.36	20.70	25,228	1,215,383	0.00	0.0	22,280
14.	SERVICE MODULE BURNOUT	12.49	44.36	20.70	24,869	1,215,383	0.00	0.0	21,410
15.	COMMAND MODULE RE-ENTRY	1,608.90	28.96	-166.02	25,831	400,000	-1.52	180.0	11,000

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) S-IC WAS SHUT DOWN 8 SEC EARLY TO PLACE SPENT BOOSTER AND LES IN A SAFE IMPACT AREA
4) PITCH-UP OF ONE-DEG/SEC FOR 10 SEC APPROXIMATES OPTIMUM TRAJECTORY, AFTER WHICH A PITCH-DOWN RATE OF 0.0885 DEG/SEC IS MAINTAINED TO S-IVB BURNOUT. A YAW PROGRAM OF ONE-DEG/SEC BETWEEN 120-150 AND BETWEEN 159 and 173.8 SEC PROVIDES AN ORBIT INCLINATION OF 90 DEG.
5) AFTER JETTISON EVENT OF 612,619 LB.
6) AFTER JETTISON EVENT OF 9,770 LB.
7) AFTER JETTISON EVENT OF 8,200 LB.
8) AFTER JETTISON EVENT OF 100,921 LB.
9) REPRESENTS UNCONSUMED S-IVB PROPELLANT (35,254 LB), S-IVB JETTISON WEIGHT (40,539 LB) AND 110,000 LB.
10) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-13

[redacted]

[redacted]

TABLE 5-20. SATURN V REFERENCE TRAJECTORIES, LOW-EARTH ORBITS
(200 N MI CIRCULAR, i = 96.5 DEG, S-IVB STACK LIMIT)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY 1) (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE 1) (DEG)	AZIMUTH 2) (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	—	6,367,048
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	12.00	28.65	-80.64	1,343	505	3.79	146.0	6,020,588
3.	SHUTDOWN OF S-IC INBOARD ENGINE	146.57	28.42	-80.50	5,948	227,908	52.88	161.8	2,135,403
4.	SHUTDOWN OF S-IC OUTBOARD ENGINES 3), BEGIN COAST	150.57	28.39	-80.48	6,215	247,244	52.65	164.6	2,043,013
5.	JETTISON OF S-IC, S-II IGNITION, BEGIN PITCH-UP MANEUVER 4), MIXTURE RATIO = 5.0	154.37	28.36	-80.47	6,121	265,764	52.01	164.6	1,430,394 5)
6.	END PITCH-UP MANEUVER, CHANGE MIXTURE RATIO TO 5.4	164.37	28.27	-80.45	6,069	313,260	50.67	166.4	1,406,098
7.	JETTISON S-IC/S-II INTERSTAGE ADAPTER SECTION	180.57	28.13	-80.42	5,951	387,588	48.64	170.5	1,352,888 6)
8.	JETTISON LAUNCH ESCAPE SYSTEM	185.57	28.08	-80.41	5,945	409,794	47.90	172.0	1,331,281 7)
9.	CHANGE MIXTURE RATIO TO 4.7	378.28	25.22	-81.06	8,633	1,008,434	14.12	199.2	814,520
10.	SHUTDOWN OF S-II, BEGIN COAST	526.19	20.99	-82.74	14,862	1,249,620	6.05	199.9	482,432
11.	JETTISON S-II, S-IVB IGNITION	530.99	20.81	-82.81	14,848	1,256,776	5.70	199.9	381,511 8)
12.	BURNOUT S-IVB, END INTEGRATED TRAJECTORY	973.23	-0.88	-89.02	25,228	1,215,772	0.00	186.5	166,617 9)
13.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER 10)	0.00	0.00	25.00	25,228	1,215,772	0.00	186.5	22,280
14.	SERVICE MODULE BURNOUT	12.49	0.00	25.00	24,869	1,215,383	0.00	186.5	21,410
15.	COMMAND MODULE RE-ENTRY	1,704.10	-66.29	-147.14	25,831	400,000	-1.52	343.7	11,000

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) S-IC WAS SHUTDOWN 8 SEC EARLY TO PLACE SPENT BOOSTER AND LES IN A SAFE IMPACT AREA
4) PITCH-UP OF ONE-DEG/SEC FOR 10 SEC APPROXIMATES OPTIMUM TRAJECTORY, AFTER WHICH A PITCH-DOWN RATE OF 0.0871 DEG/SEC IS MAINTAINED TO S-IVB BURNOUT. A YAW PROGRAM OF ONE-DEG/SEC BETWEEN 120-150 AND BETWEEN 159 AND 183.8 SEC PROVIDES AN ORBIT INCLINATION OF 96.5 DEG.
5) AFTER JETTISON EVENT OF 612,619 LB.
6) AFTER JETTISON EVENT OF 9,770 LB.
7) AFTER JETTISON EVENT OF 8,200 LB.
8) AFTER JETTISON EVENT OF 100,921 LB.
9) REPRESENTS UNCONSUMED S-IVB PROPELLANT (16,078 LB), S-IVB JETTISON WEIGHT (40,539 LB) AND 110,000 LB.
10) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-15.

[redacted]

5.2 Earth Synchronous Orbits

5.2.1 Inclined

The synchronous orbit payload capabilities of the Saturn V launch vehicle have been evaluated for several ascent modes. All launches were due east from ETR, resulting in an orbital inclination of 28.°5.

A coast period of 20 minutes is allotted between S-IVB and Service Module operation to simulate the transposition and docking maneuver. In some cases, such as when the S-IVB is burned at apogee, this necessitates waiting a full period and then igniting the SM. While this imposes no performance penalties, other systems considerations may be involved.

The portion of the Service Module propellant usable during the ascent phase will be determined largely by the deboost requirements. The deboost velocity requirement is about 4870 fps for an equatorial landing and is increased by 10 percent for performance contingencies and another 100 fps for a midcourse maneuver. Assuming the payload is left in orbit the following CSM weight breakdown will be required for deboost.

Command Module	11,000 lb
Service Module	10,200
Propellant	15,250
Initial Deboost	36,450 lb

This deboost maneuver results in a re-entry flight path angle of -6.4 degrees at an altitude of 400,000 feet.

The flight mode which offers the best payload is given in Table 5-21. First burn of the S-IVB establishes the transfer orbit to the altitude of the 24-hour orbit. Near apogee, the S-IVB re-ignites and burns the remainder of its propellant, i.e., down to but not including the flight performance reserve. After the docking maneuver, the CSM has coasted past apogee into a region where an attempt to circularize is inefficient. A waiting period, until the CSM is near apogee again, is necessary. The period of this waiting orbit is 13.65 hours and a perigee results of about 5106 n.mi. Apogee is still at synchronous orbit altitude. At apogee, the CSM ignites accelerating the payload to circular velocity The total weight in the synchronous orbit is 69,689 pounds. Of this weight 36,450 is required for deboost and 5,310 pounds for crew/45 day mission subsistence. The remainder, 27,929 pounds, represents net payload.

The possibility of increasing net payload by using LEM descent stage or ascent stage propulsion was considered.

The utilization of the LEM descent stage to accomplish the synchronous orbit involves either its use for orbit circularization or in implementing the deboost process. It can be shown that the former results in no appreciable net payload gain. The latter would involve jettisoning the service Module in orbit (the full systems implications of this action must be investigated); however, the experiment payload and the LEM Lab must be deboosted in a practical configuration. This action would then result in the following deboost stage breakdown.

Command Module	11,000 lb
Descent Stage	3,500 lb
LEM Lab & Experiments	16,600 lbs
Propellant (Descent Stage)	23,080 lbs
	54,180 lbs

[redacted]

TABLE 5-21. SATURN V REFERENCE TRAJECTORIES, SYNCHRONOUS ORBITS (i = 28.5 DEG)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1] (DEG)	AZIMUTH[2] (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	-	6,350,851
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	12.00	28.65	-80.64	1,343	512	3.84	90.0	6,004,390
3.	SHUTDOWN OF S-IC INBOARD ENGINE	154.57	28.64	-79.85	8,648	191,838	20.10	90.7	1,888,232
4.	SHUTDOWN OF S-IC OUTBOARD ENGINES, BEGIN COAST	158.57	28.64	-79.76	9,093	203,885	22.94	90.7	1,795,842
5.	JETTISON OF S-IC, S-II IGNITION, BEGIN PITCH-UP MANEUVER,[3] MIXTURE RATIO = 5.0	162.37	28.64	-79.68	9,053	215,281	18.99	90.8	1,414,198[4]
6.	END PITCH-UP MANEUVER, CHANGE MIXTURE RATIO TO 5.4	172.37	28.64	-79.45	9,188	243,846	17.58	91.0	1,389,902
7.	JETTISON S-IC/S-II INTERSTAGE ADAPTER SECTION	188.57	28.63	-79.07	9,459	287,004	15.67	91.2	1,336,692[5]
8.	JETTISON LAUNCH ESCAPE SYSTEM	193.57	28.63	-78.95	9,550	299,613	15.11	91.3	1,315,084[6]
9.	CHANGE MIXTURE RATIO TO 4.7	386.28	28.33	-72.77	15,302	572,189	2.15	94.9	798,324
10.	SHUTDOWN OF S-II, BEGIN COAST	534.19	27.51	-65.19	22,446	606,342	0.52	98.9	466,246
11.	JETTISON S-II, S-IVB IGNITION	538.99	27.47	-64.89	22,445	607,204	0.44	99.0	365,315[7]
12.	END INTEGRATED TRAJECTORY (100 N MI CIRCULAR PARKING ORBIT)	694.13	25.59	-54.69	25,582	607,293	0.00	104.1	289,930
13.	CONTINUATION OF S-IVB FOR PERIGEE OVERSPEED	694.13	25.59	-54.69	25,582	607,293	0.00	103.3[1]	289,930[8]
14.	BURNOUT OF S-IVB, COAST TO APOGEE	959.77	25.59	-54.69	33,657	607,293	0.00	103.3	160,853
15.	RE-START OF S-IVB FOR APOGEE MANEUVER	19,379.69	-25.59	46.43	5,234	19,329[9]	0.00	76.7	156,913[10]
16.	BURNOUT OF S-IVB	19,927.91	-25.59	46.43	7,454	19,329	0.00	76.7	133,483
17.	SERVICE MODULE IGNITION	69,065.37[12]	-25.59	-158.26	7,454	19,329	0.00	76.7	92,943
18.	SERVICE MODULE BURNOUT (CIRCULAR ORBIT)	69,399.75	-25.59	-158.26	10,089	19,329	0.00	76.7	69,689
19.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER[13]	0.00	-28.66	100.00	10,089	19,329	0.00	90.0	36,450[11]
20.	SERVICE MODULE BURNOUT	216.00	-28.66	100.00	5,219	19,329	0.00	90.0	21,410
21.	COMMAND MODULE RE-ENTRY	13,672.90	27.63	-174.71	33,840	65.83	-6.40	97.8	11,000

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) PITCH-UP OF ONE-DEG/SEC FOR 10 SEC APPROXIMATES OPTIMUM TRAJECTORY, AFTER WHICH A PITCH-DOWN RATE OF 0.0993 DEG/SEC IS MAINTAINED TO FIRST S-IVB BURNOUT
4) AFTER JETTISON EVENT OF 381,645 LB
5) AFTER JETTISON EVENT OF 9,770 LB
6) AFTER JETTISON EVENT OF 8,200 LB
7) AFTER JETTISON EVENT OF 100,921 LB
8) ASSUME NO COAST
9) NAUTICAL MILES
10) 3940 LB LOST DUE TO BOILOFF DURING COAST TO APOGEE
11) THE DIFFERENCE BETWEEN 18 AND 19 REPRESENTS THE PAYLOAD LEFT ON ORBIT OR CONSUMED DURING STAY
12) ALLOWS ONE ORBIT PRIOR TO S/M OPERATION: PERIOD = 13.649 HRS, PERIGEE = 5106.3 N MI
13) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-22.

[redacted]

Thus, this method yields not only a substantial payload reduction but also requires descent stage propulsion beyond the available tankage. If by some means the experiment payload could be jettisoned prior to deboost, then the method would result in a net payload gain of approximately 10,000 pounds and propulsion requirements within tankage size.

The utilization of LEM ascent stage propulsion (LEM LAB is provided with a propulsive capability) involves the same considerations as above. If such propulsion were used for deboost and the experiment payload could be jettisoned prior to deboost together with the Service Module, then a net payload gain of approximately 9000 pounds would result. However, propulsion requirements (11,700 pounds) would greatly exceed ascent stage tankage.

The trajectory of a descent profile from an inclined synchronous orbit is given in Table 5-22.

TABLE 5-22. DESCENT FROM SYNCHRONOUS ORBIT; i = 28.5 DEGREES

TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1)] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1)] (DEG)	AZIMUTH[1)] (DEG)
0	-28.66	100.00	5,181	117,432,840	0.00	90.0
2000	-28.57	96.55	5,334	116,347,465	-11.77	87.7
4000	-28.30	93.24	5,786	113,062,856	-22.39	85.3
6000	-27.80	90.24	6,527	107,489,442	-31.21	82.8
8000	-27.00	87.78	7,564	99,462,848	-38.08	80.1
9000	-26.45	86.84	8,208	94,451,238	-40.81	78.6
10,000	-25.77	86.19	8,952	88,715,400	-43.09	77.1
11,000	-24.90	85.91	9,817	82,196,735	-44.92	75.4
12,000	-23.79	86.12	10,836	74,820,722	-46.30	73.6
13,000	-22.31	87.02	12,060	66,491,717	-47.16	71.6
14,000	-20.30	88.92	13,574	57,086,157	-47.40	69.4
15,000	-17.38	92.38	15,533	46,445,993	-46.76	66.0
16,000	-12.82	98.55	18,232	34,386,005	-44.70	64.3
17,000	-4.69	110.33	22,317	20,806,088	-39.68	61.8
18,000	12.20	138.07	29,066	6,645,655	-26.55	64.0
18,672.9	27.63	-174.71	33,833	400,000	-6.28	82.2

1) INERTIAL QUANTITIES

5.2.2 Equatorial

The equatorial synchronous orbit capabilities of the Saturn V were evaluated for several ascent modes including both 2 and 3 S-IVB burn periods. Again, a due east launch was assumed from ETR. A description of some of the more significant flight modes considered is given here along with a mission profile of the maximum payload configuration.

The performance of an arrangement with the constraint that the S-IVB be burned only once was computed. In this capacity, the S-II injects an offloaded S-IVB onto a low altitude parking orbit. The altitude of the parking orbit for this and all other cases was 100 n.mi. The offloaded S-IVB coasts to the first equatorial crossing and begins its burning phase. If the S-IVB accelerates to the perigee conditions required for apogee to be at synchronous altitude, the Service Module must provide the remaining impulse to circularize the orbit and change the plane by 28.5 degrees. This requires more SM propellant than is available. If, however, the S-IVB expends some of its excess capability and provides a portion of the plane change at perigee the velocity required at apogee is reduced as well as the payload and to the point where the Service Module can perform the maneuver. In this case a plane change of about 18 degrees is made at perigee and the remainder at apogee. The burnout weight is 31,870 pounds which is less than the CSM weight requirement for deboost. A payload of about 5300 pounds is possible if the LEM Descent Stage is used for deboost. This, however, is barely enough for the 45-day mission. Thus, the net payload, if any, is marginal.

For a two-burn S-IVB, the S-II stage again injects an offloaded S-IVB onto a 100 n.mi. parking orbit. The S-IVB starts at the equator establishing the perigee conditions for the desired apogee. After coasting to apogee which also occurs on the equator the S-IVB restarts to circularize the orbit and simultaneously turn equatorial. The Service Module carried only enough propellant for deboost to an equatorial landing. The total weight in the synchronous orbit is 82,330 pounds of which 40,540 is the spent S-IVB. This leaves 41,790 pounds which again is just the requirement for the 45-day stay and deboost.

The most efficient way of arranging the two-burn S-IVB is to carry the Service Module, fully loaded, and combine the second burn of the S-IVB (at apogee) with the first burn of the Service Module. This gives a net payload of 3600 pounds.

Many cases were investigated wherein the S-II was used to loft the S-IVB to varying apogee altitudes. It was found that the most efficient incorporated a low altitude parking orbit.

Considerable increases in payload are realized if a third-burn capability is added to the S-IVB. In such a case the first burnout of the S-IVB establishes perigee and changes the orbit plane about two degrees. At apogee the S-IVB starts for its third burn and consumes the remainder of the usable propellant. At this point a waiting orbit is necessitated by the transposition and docking maneuver. Its period is 14.097 hours and has a perigee altitude of 5782.6 n.mi. At the next apogee, the Service Module ignites to finish the plane change and circularize the orbit. Total burnout weight is 61,334 pounds which gives a net payload of 19,573 pounds. A summary of this mission profile is given in Table 5-23. The descent phase for an equatorial landing site at the longitude of the Apollo impact area is given in Table 5-24.

An analysis was made to evaluate the performance penalties associated with leaving the equatorial orbit for the nominal Apollo landing sites, i.e., at a latitude of ± 20 degrees. To do this, an assessment was made of the velocity requirements of various descent modes. All modes assumed the original orbit to be circular, synchronous and equatorial and positioned (in longitude) for descent. The flight path at 400,000 feet was -6.4 degrees. Entry body aerodynamics were not considered.

<u>Mode 1.</u> ΔV_1 is applied opposite the circular velocity, V_c, vector. No plane change and impact is on the equator; (ΔV_1 = 4870 fps; total ΔV = 4870 fps).

<u>Mode 2.</u> ΔV_1 is applied to V_c in such a direction as to change only the orbit inclination; ΔV_2 is applied opposite V_c at the appropriate time for landing; (ΔV_1 = 3465 fps; ΔV_2 = 4870 fps; total ΔV = 8335 fps).

CONFIDENTIAL

Table 5-23. SATURN V REFERENCE TRAJECTORIES, SYNCHRONOUS ORBITS (i = 0 DEG)

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1] (DEG)	AZIMUTH[2] (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	-	6,341,035
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	12.00	28.65	-80.64	1,343	517	3.87	90.0	5,994,575
3.	SHUTDOWN OF S-IC INBOARD ENGINE	154.57	28.64	-79.84	8,693	192,289	20.01	90.7	1,878,416
4.	SHUTDOWN OF S-IC OUTBOARD ENGINES, BEGIN COAST	158.57	28.64	-79.75	9,141	204,345	19.52	90.8	1,786,027
5.	JETTISON OF S-IC, S-II IGNITION, BEGIN PITCH-UP MANEUVER,[3] MIXTURE RATIO = 5.0	162.37	28.64	-79.67	9,101	215,750	18.90	90.8	1,404,382[4]
6.	END PITCH-UP MANEUVER, CHANGE MIXTURE RATIO TO 5.4	172.37	28.64	-79.44	9,238	244,342	17.50	91.0	1,380,086
7.	JETTISON S-IC/S-II INTERSTAGE ADAPTER SECTION	188.57	28.63	-79.06	9,514	287,565	15.61	91.2	1,326,876[5]
8.	JETTISON LAUNCH ESCAPE SYSTEM	193.57	28.63	-78.94	9,605	300,198	15.05	91.3	1,305,268[6]
9.	CHANGE MIXTURE RATIO TO 4.7	386.28	26.32	-72.70	15,439	574,130	2.14	94.9	788,508
10.	SHUTDOWN OF S-II, BEGIN COAST	534.19	27.49	-65.04	22,731	606,632	0.44	98.9	456,420
11.	JETTISON S-II, S-IVB IGNITION	538.99	27.45	-64.73	22,730	607,358	0.36	99.1	355,499[7]
12.	BURNOUT OF S-IVB, END INTEGRATED TRAJECTORY, COAST TO EQUATOR	677.30	25.79	-55.57	25,582	607,831	0.00	103.6	288,293
13.	RE-START OF S-IVB FOR PERIGEE OVERSPEED (AND 2-DEG PLANE CHANGE)	1,631.40	0.00	2.49	25,582	607,831	0.00	118.5[1]	288,033[8]
14.	BURNOUT OF S-IVB, COAST TO APOGEE	1,896.34	0.00	2.49	33,657	607,831	0.00	116.5	159,052
15.	RE-START OF S-IVB FOR APOGEE MANEUVER	20,316.76	0.00	98.68	5,234	19,329[9]	0.00	63.5	155,112[10]
16.	BURNOUT OF S-IVB	20,378.51	0.00	98.68	7,661	19,329	0.00	76.8	125,131
17.	SERVICE MODULE IGNITION	71,616.74[12]	0.00	-112.78	7,661	19,329	0.00	76.8	84,592
18.	SERVICE MODULE BURNOUT (CIRCULAR ORBIT)	72,006.65	0.00	-112.78	10,089	19,329	0.00	90.0	61,334
19.	SERVICE MODULE IGNITION FOR DEBOOST MANEUVER[13]	0.00	0.00	83.30	10,089	19,329	0.00	90.0	36,450[11]
20.	SERVICE MODULE BURNOUT	216.00	0.00	83.30	5,219	19,329	0.00	90.0	21,410
21.	COMMAND MODULE RE-ENTRY	13,668.50	0.00	170.00	33,840	65.83[4]	-6.40	90.0	11,000

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) PITCH-UP OF ONE-DEG/SEC FOR 10-SEC APPROXIMATES OPTIMUM TRAJECTORY, AFTER WHICH A PITCH-DOWN RATE OF 0.1021 DEG/SEC IS MAINTAINED TO FIRST S-IVB BURNOUT
4) AFTER JETTISON EVENT OF 381,645 LB
5) AFTER JETTISON EVENT OF 9,770 LB
6) AFTER JETTISON EVENT OF 8,200 LB
7) AFTER JETTISON EVENT OF 100,921 LB
8) 260 LB LOST DUE TO BOILOFF DURING COAST TO EQUATOR
9) NAUTICAL MILES
10) 3940 LB LOST DUE TO BOILOFF DURING COAST TO APOGEE
11) THE DIFFERENCE BETWEEN 18 AND 19 REPRESENTS THE PAYLOAD LEFT ON ORBIT OR CONSUMED DURING STAY
12) ALLOWS ONE ORBIT PRIOR TO S/H OPERATION: PERIOD = 14.097 HRS, PERIGEE = 5,782.6 N MI
13) THE TIME HISTORY OF A DESCENT PROFILE FOR THIS MISSION IS GIVEN IN TABLE 5-2

CONFIDENTIAL

TABLE 5-24. DESCENT FROM SYNCHRONOUS ORBIT; i = 0 DEGREES

TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY[1] (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE[1] (DEG)	AZIMUTH[1] (DEG)
0	0.00	83.30	5181	117,432,842	0.00	90.0
2000	0.00	79.25	5333	116,347,668	-11.77	90.0
4000	0.00	73.35	5785	113,063,674	-22.39	90.0
6000	0.00	71.76	6526	107,491,338	-31.21	90.0
8000	0.00	68.69	7563	99,466,383	-38.08	90.0
9000	0.00	64.47	8207	94,455,855	-40.81	90.0
10,000	0.00	66.53	8951	88,721,309	-43.09	90.0
11,000	0.00	65.98	9815	82,204,194	-44.93	90.0
12,000	0.00	65.96	10,834	74,830,045	-46.30	90.0
13,000	0.00	66.67	12,057	66,503,295	-47.16	90.0
14,000	0.00	68.50	13,571	57,100,484	-47.40	90.0
15,000	0.00	72.11	15,529	46,463,657	-46.77	90.0
16,000	0.00	78.90	18,226	34,407,516	-44.71	90.0
17,000	0.00	92.40	22,308	20,830,545	-39.70	90.0
18,000	0.00	124.07	29,055	6,660,801	-26.60	90.0
18,668.5	0.00	170.00	33,826	400,000	- 6.52	90.0

1) INERTIAL QUANTITIES

Mode 3. ΔV_1 is applied to V_c such that the resulting orbit has a perigee of 200 n.mi. and the orbit inclination is changed by 20 degrees. After coasting to perigee (which also occurs on the equator), ΔV_2 is applied opposite the velocity vector to provide circular velocity at that altitude. ΔV_3 is applied opposite V_c at the appropriate time for landing; (ΔV_1 = 5422 fps; ΔV_2 = 7900 fps; ΔV_3 = 357 fps; total ΔV = 13,679 fps).

Mode 4. ΔV_1 is applied opposite V_c as in Mode 1. After coasting to a point 90 degrees from perigee (true anomaly), the orbit inclination can be changed by applying ΔV_2 to the velocity vector in the proper direction; (ΔV_1 = 4870 fps; ΔV_2 = 8400 fps; total ΔV = 13,270 fps).

Thus, the descent from an equatorial synchrnous orbit to a 20 degree latitude is a serious constraint with all sorts of system implications. If the technique of Mode 2 and SM propulsion only are used, the net payload is reduced from 19,574 pounds to 2,800 pounds, and SM propulsion requirements increase from 38,300 to 55,000 pounds. Additional tankage is then required, decreasing net payload still further. On the other hand, Mode 2, SM propulsion and LEM propulsion may be used and while easing the above problem somewhat off-equatorial landings still represent sizable decreases in payload. It should be emphasized that although command module lifting re-entry will mitigate these problems, synchronous equatorial orbit performance under the proviso of Apollo site landing requires careful detail analysis with actual and practical spacecraft configurational data as inputs.

5.3 Lunar Orbital Missions

A set of lunar orbit photographic survey studies were reported in References 5,6. The primary purpose of these studies were to establish a set of orbital techniques which would best satisfy the mission objectives. Both low inclination and lunar polar orbits were involved. The required orbital techniques will not be repeated here; however, the vehicle weight histories for various configurations and flight times are indicated in Tables 5-25 to 5-27. It is to be noted that the accomplishment of the 7 day total transit Scientific Survey mission (lunar polar orbit) by SM propulsion only involves propulsion requirements exceeding the available tankage. Table 5-28 indicates the mission profile for the data of configuration 1, Table 5-27.

Table 5-25. Weight Histories, Site Certification Mission (6 Day Total Transit)

Configuration 1 - Camera Module Jettisoned Prior to Return Injection

Configuration 2 - Camera Module Jettisoned Prior to Return Midcourse Correction

Both Configurations - SM Propulsion only

Item	Weight, Lb. Config. 1	Config. 2
Gross Injected Weight	65,517	68,907
Less Adapter	- 3,700	- 3,700
Net Injected Weight	61,817	65,207
Less Trans-Lunar Midcourse Correction Propellant	- 1,856	- 1,956
Vehicle at End Midcourse Correction	59,961	63,251
Less Deboost Propellant	-16,375	-17,280
Vehicle in Lunar Orbit	43,586	45,971
Less Orbital Maneuver Propellant	- 9,285	- 9,700
Less Camera Module	- 5,000	-
Vehicle at Start Trans-Earth Injection	29,301	36,181
Less Trans-Earth Injection Propellant	- 8,000	- 9,880
Less Camera Module	-	- 5,000
Vehicle in Trans-Earth Trajectory	21,301	21,301
Less Return Midcourse Propellant	- 639	- 639
Vehicle Prior to SM Separation	20,662	20,662
TOTAL SM PROPELLANT	36,155	39,545

Table 5-26. Weight Histories, Scientific Survey Mission (7 Day Total Transit)

Configuration 1 - Camera Module Jettisoned Prior to Return Injection

Configuration 2 - Camera Module Jettisoned Prior to Return Midcourse Correction

Both Configurations - SM Propulsion only

Item	Weight, Lb. Config. 1	Config. 2
Gross Injected Weight	80,076	85,087
Less Adapter	- 3,700	- 3,700
Net Injected Weight	77,376	81,387
Less Trans-Lunar Midcourse Correction Propellant	- 2,321	- 2,441
Vehicle at End Midcourse Correction	75,055	78,946
Less Deboost Propellant	-20,490	-21,552
Vehicle in Lunar Orbit	54,565	57,394
Less Orbital Maneuver Propellant	-11,622	-12,225
Less Camera Module	- 5,000	-
Vehicle at Start Trans-Earth Injection	37,943	45,169
Less Trans-Earth Injection Propellant	-10,358	-12,331
Less Camera Module	-	- 5,000
Vehicle in Trans-Earth Trajectory	27,585	27,838
Less Return Midcourse Propellant	- 827	- 835
Vehicle Prior to SM Separation	26,758	27,003
TOTAL SM PROPELLANT	45,618	49,384

Table 5-27. Weight Histories, Scientific Survey Mission

Configuration 1 - Service Module + Command Module + Camera Module, with Camera Module Jettisoned Prior to Trans-Earth Injection, 9-1/2 Day Total Transit

Configuration 2 - Service Module + Command Module + Camera Module + Modified LEM Descent Stage, with the Camera Module and Modified LEM Descent Stage Jettisoned Prior to Trans-Earth Injection, 7 Day Total Transit

Both SM and LEM Descent Stage Provide Propulsion

Item	Weight, Lb. Config. 1	Weight, Lb. Config. 2
Gross Injected Weight	75,704	85,302
Less Adapter	- 3,700	- 3,700
Net Injected Weight	72,004	81,602
Less Trans-Lunar Midcourse Correction Propellant	- 2,160	- 2,446
Vehicle at End Midcourse Correction	69,844	79,156
Less Deboost Propellant	-17,531	-21,609
Vehicle in Lunar Orbit	52,313	57,547
Less Orbital Maneuver Propellant	-11,143	-12,486
Less Camera Module	- 5,000	- 5,000
Less Modified LEM Descent Stage	-	- 2,800
Vehicle at Start Trans-Earth Injection	36,170	37,261
Less Trans-Earth Injection Propellant	- 9,079	-10,170
Vehicle in Trans-Earth Trajectory	27,091	27,091
Less Return Midcourse Propellant	- 813	- 813
Vehicle Prior to SM Separation	26,278	26,278
TOTAL PROPELLANT	SM - 40,726	SM - 35,038
		LEM - 12,486

TABLE 5-28

SATURN V SCIENTIFIC SURVEY MISSION (CONFIGURATION 1, TABLE 5-27)

PARAMETER	
LUNAR OPERATIONS	
1. TIME OF INJECTION (MIN GMT)	1224.4 JUNE 28, 69
2. WEIGHT AT INJECTION (LB)	72,004
3. TIME OF PERICYNTHION (MIN GMT)	614.4 JULY 03, 69
4. TRANSEARTH HYPERBOLIC INCLINATION (DEG)	87.046
5. PERICYNTHION ALTITUDE (N MI)	79.095
6. PARKING ORBIT INCLINATION (DEG)	90
7. P. O. NODE AT PERICYNTHION (DEG)	88.620
8. DEBOOST PLANE CHANGE (DEG)	6.638
9. MIDCOURSE VELOCITY INCREMENT (FPS)	370
10. DEBOOST VELOCITY INCREMENT (FPS)	2818.6
11. WEIGHT OF SM PROPELLANT FOR (9 AND 10)	19,231
12. CSM WEIGHT AFTER LOI (LBS)	52,773
13. PARKING ORBIT DURATION (HOURS)	674
14. WEIGHT EXPENDED DURING WAIT (LB)	11,191
15. CAMERA WEIGHT LEFT IN LUNAR ORBIT (LB)	5,000
TRANSEARTH TRAJECTORY	
1. TIME OF INJECTION (MIN GMT)	720 JULY 31, 69
2. CSM WEIGHT PRIOR TO TRANSEARTH INJECTION (LB)	36,582
3. TEI PLANE CHANGE (DEG)	7.746
4. TEI VELOCITY INCREMENT (FPS)	2857
5. MIDCOURSE VELOCITY CORRECTION (FPS)	250
6. WEIGHT OF SM PROPELLANT FOR 4 AND 5 (LB)	9722
7. CSM WEIGHT AFTER TEI (LB)	26,860
8. TRANSEARTH FLIGHT TIME (HOURS)	110
9. LANDING SITE	SAMOA
10. RETURN INCLINATION (DEG)	40
11. RE-ENTRY MANEUVER ANGLE (DEG)	33
12. TIME OF LANDING (MIN GMT)	111.7 AUG 5, 69

5.4 Lunar Landing Missions

An Extended Stay Lunar Exploration Mission involving consecutive landings of two vehicles (LEM Shelter, LEM Taxi) at a selected lunar site for extended periods has been studied and reported in Reference 6. The primary purpose of the study was to evolve trajectory techniques which maximized lunar accessibility in terms of available sites and stay time, and still be within the basic Apollo constraints. The details will not be repeated here, but it was found necessary, in general, to violate the continuous abort constraint.

Tables 5-29 thru 5-32 indicate the LEM Shelter and LEM Taxi vehicle mission profiles for two arbitrarily selected lunar landing sites (0° lat, -80° long) and (45° lat, -80° long). The discrepancy between the LEM Shelter and LEM Taxi dates exist because of the available data that could be immediately utilized. Nevertheless, representative trajectories are shown.

[redacted]

TABLE 5-29. SATURN V LUNAR EXPLORATION MISSION, TRANSLUNAR TRAJECTORY, LEM SHELTER OR LEM TAXI

	TRAJECTORY EVENT	TIME (SEC)	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY 1) (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE 1) (DEG)	AZIMUTH 2) (DEG)	VEHICLE WEIGHT (LB)
1.	LIFTOFF	0.00	28.65	-80.64	1,340	0	0.00	-	6,000,000
2.	END VERTICAL RISE, BEGIN ZERO-LIFT FLIGHT	12.00	28.65	-80.64	1,344	637	4.74	90.0	5,658,548
3.	SHUTDOWN OF S-IC INBOARD ENGINE	146.08	28.64	-79.89	8,512	189,638	21.50	90.6	1,843,364
4.	SHUTDOWN OF S-IC OUTBOARD ENGINES, BEGIN COAST	150.08	28.64	-79.81	8,960	202,307	21.03	90.7	1,752,310
5.	JETTISON OF S-IC, S-II IGNITION, BEGIN PITCH-UP MANEUVER 3)	153.88	28.64	-79.72	8,917	214,321	20.40	90.8	1,389,105 4)
6.	END PITCH-UP MANEUVER, BEGIN PITCH-DOWN RATE	165.42	28.64	-79.47	9,060	249,036	18.78	90.9	1,362,013
7.	JETTISON S-IC/S-II INTERSTAGE ADAPTER SECTION	183.88	28.63	-79.05	9,307	300,490	16.60	91.2	1,309,232 5)
8.	JETTISON LAUNCH ESCAPE SYSTEM	188.88	28.63	-78.94	9,381	313,613	16.03	91.3	1,290,895 6)
9.	SHUTDOWN OF S-II, INITIATE COAST	547.83	27.47	-64.94	22,414	609,832	0.44	99.0	448,300
10.	JETTISON S-II, S-IVB IGNITION	552.63	27.43	-64.64	22,413	610,540	0.36	99.2	360,445 7)
11.	BURNOUT OF S-IVB, BEGIN PARKING ORBIT, END INTEGRATED TRAJECTORY	712.53	25.46	-54.16	25,581	607,629	0.00	104.4	285,372
12.	RE-START OF S-IVB	9367.95			25,581	607,629	0.00		283,937 8)
13.	BURNOUT OF S-IVB, TRANSLUNAR INJECTION 9)	9691.31			35,592	911,415	6.60		132,125
14.	BEGIN TRANSLUNAR COAST	9691.31			35,592	911,415	6.60		91,585 10)

1) INERTIAL QUANTITIES
2) RELATIVE QUANTITIES
3) PITCH-UP OF ONE-DEG/SEC FOR 11.54 SEC APPROXIMATES OPTIMUM TRAJECTORY, AFTER WHICH A PITCH-DOWN RATE OF 0.1003 DEG/SEC IS MAINTAINED TO PARKING ORBIT
4) AFTER JETTISON EVENT OF 363,205 LB
5) AFTER JETTISON EVENT OF 9,450 LB
6) AFTER JETTISON EVENT OF 6,600 LB
7) AFTER JETTISON EVENT OF 87,855 LB
8) 1435 LB LOST DUE TO BOILOFF DURING PARKING ORBIT
9) INJECTION CONDITIONS ARE FOR 110-HOUR TRANSLUNAR FLIGHT TIME
10) AFTER JETTISON EVENT OF 40,540 LB

[redacted]

TABLE 5-30. SATURN V LUNAR EXPLORATION MISSION, LEM SHELTER

	PARAMETER	LANDING SITE LOCATION (LAT., LONG.) 0°, -80°	45°, -80°
	LUNAR OPERATIONS		
1.	TIME OF INJECTION (MIN GMT)	903.4 JUL 26,69	903.5 JUL 26,69
2.	CSM WEIGHT AT INJECTION (LB)	90,223	90,259
3.	TIME OF PERICYNTHION (MIN GMT)	742.4 JUL 30,69	504.9 JUL 31,69
4.	TRANSLUNAR HYPERBOLIC INCLINATION (DEG)	171.40	95.40
5.	PERICYNTHION ALTITUDE (N MI)	79.80	79.80
6.	PARKING ORBIT INCLINATION (DEG)	171.40	95.40
7.	P. O. NODE AT PERICYNTHION (DEG)	283.1	287.90
8.	DEBOOST PLANE CHANGE (DEG)	0	0
9.	MIDCOURSE VELOCITY INCREMENT (FPS)	370	370
10.	DEBOOST VELOCITY INCREMENT (FPS)	2,715	2,720
11.	WEIGHT OF SM PROPELLANT FOR 9 and 10 (LB)	23,720	23,718
12.	CSM WEIGHT AFTER INSERTION (LUNAR P.O.)	66,503	66,541
13.	PARKING ORBIT WAITING TIME (HOURS)	5.25	5.30
14.	WEIGHT EXPENDED DURING WAIT (LB)	0	0
15.	LEM PLANE CHANGE AT LIFTOFF (DEG)	0	0
	LEM DESCENT/ASCENT		
1.	START WEIGHT OF HOHMANN TRANSFER	33,500	
	TRANSEARTH TRAJECTORY		
1.	TIME OF INJECTION (MIN GMT)	733.7 JUL 31,69	792 AUG 1,69
2.	CSM WEIGHT PRIOR TO TRANSEARTH INJECTION (LB)	33,003	33,041
3.	TRANSEARTH INJECTION PLANE CHANGE (DEG)	5.20	5.80
4.	TRANSEARTH VELOCITY INCREMENT (FPS)	2,768	2,772
5.	MIDCOURSE VELOCITY AND CONTINGENCIES (FPS)	815	815
6.	WEIGHT OF SM PROPELLANT FOR 4 AND 5 (LB)	9,933	9,871
7.	CSM WEIGHT AFTER TRANSEARTH INJECTION (LB)	23,070	23,170
8.	TRANSEARTH FLIGHT TIME (HOURS)	109.633	109.45
9.	LANDING SITE	HAWAII	HAWAII
10.	RETURN INCLINATION (DEG)	40	40
11.	RE-ENTRY MANEUVER ANGLE (DEG)	33	33
12.	TIME OF LANDING (MIN GMT)	111.7 AUG 5,69	159.6 AUG 6,69

TABLE 5-31. SATURN V LUNAR EXPLORATION MISSION, LEM TAXI

PARAMETER	LANDING SITE LOCATION (LAT., LONG.) 0°, -80°	45°, -80°
LUNAR OPERATIONS		
1. TIME OF INJECTION (MIN GMT)	193.3 FEB 25,72	193.3 FEB 25,72
2. CSM WEIGHT AT INJECTION (LB)	91,585	91,585
3. TIME OF PERICYNTHION (MIN GMT)	1033.3 FEB 29,72	1033.3 FEB 29.72
4. TRANSLUNAR HYPERBOLIC INCLINATION (DEG)	128.28	91.93
5. PERICYNTHION ALTITUDE (N MI)	79.24	79.97
6. PARKING ORBIT INCLINATION (DEG)	174.39	91.12
7. P. O. NODE AT PERICYNTHION (DEG)	287.87	284.08
8. DEBOOST PLANE CHANGE (DEG)	5.83	2.77
9. MIDCOURSE VELOCITY INCREMENT (FPS)	370	370
10. DEBOOST VELOCITY INCREMENT (FPS)	2,712	2,643
11. WEIGHT OF SM PROPELLANT FOR 9 AND 10(LB)	24,143	23,682
12. CSM WEIGHT AFTER INSERTION (LUNAR P.O.)	67,442	67,903
13. PARKING ORBIT WAITING TIME (HOURS)	0	19
14. WEIGHT EXPENDED DURING WAIT (LB)	0	48
15. LEM PLANE CHANGE AT LIFTOFF (DEG)	1.00	2.00
LEM DESCENT/ASCENT		
1. START WEIGHT OF HOHMANN TRANSFER	29,980	
2. START WEIGHT OF FINAL DESCENT	29,647	
3. WEIGHT AT TOUCHDOWN	15,236	
4. WEIGHT EXPENDED DURING STAY	4,736	
5. LIFTOFF WEIGHT	10,500	
6. WEIGHT AFTER RENDEZVOUS	5,124	
7. JETTISON WEIGHT	5,124	
TRANSEARTH TRAJECTORY		
1. TIME OF INJECTION (MIN GMT)	180.0 MAR 15,72	1400.0 MAR 15,72
2. CSM WEIGHT PRIOR TO TRANSEARTH INJECTION (LB)	37,462	37,875
3. TRANSEARTH INJECTION PLANE CHANGE (DEG)	0.25	3.32
4. TRANSEARTH VELOCITY INCREMENT (FPS)	2,727	2,746
5. MIDCOURSE VELOCITY AND CONTINGENCIES (FPS)	815	815
6. WEIGHT OF SM PROPELLANT FOR 4 AND 5 (LB)	11,108	11,296
7. CSM WEIGHT AFTER TRANSEARTH INJECTION (LB)	26,354	26,579
8. TRANSEARTH FLIGHT TIME (HOURS)	104.3	109.1
9. LANDING SITE	HAWAII	HAWAII
10. RETURN INCLINATION (DEG)	40	40
11. REENTRY MANEUVER ANGLE (DEG)	33	33
12. TIME OF LANDING (MIN GMT)	680.4 MAR 19,72	744.0 MAR 20,72

CONFIDENTIAL

TABLE 5-32. TRANSEARTH FREE-FLIGHT; 109.17 HOURS, LEM SHELTER OR TAXI

TIME DAY	HR	MIN	GEODETIC LATITUDE (DEG)	LONGITUDE (DEG)	VELOCITY 1) (FT/SEC)	ALTITUDE (FT)	FLIGHT PATH ANGLE 1) (DEG)	AZIMUTH (DEG)
0	0	0.0 2)	12.54	-139.51	5789	1,310,439,232	57.36	290.7
0	3	0.0	12.84	174.69	931	1,308,978,240	-58.65	296.5
0	6	0.0	12.89	129.48	888	1,299,834,128	-88.09	330.4
0	12	0.0	12.84	39.37	1074	1,279,042,432	-75.39	111.0
0	18	0.0	12.72	- 50.56	1267	1,254,838,336	-71.93	112.2
0	24	0.0	12.56	-140.41	1453	1,226,930,512	-71.35	112.7
1	6	0.0	12.36	129.81	1655	1,195,065,104	-71.68	112.9
1	12	0.0	12.14	40.10	1860	1,158,985,616	-72.30	113.2
1	18	0.0	11.89	- 49.56	2079	1,118,404,720	-73.01	113.4
1	24	0.0	11.62	-139.16	2314	1,072,982,104	-73.69	113.5
2	6	0.0	11.30	131.32	2571	1,022,301,232	-74.32	113.7
2	12	0.0	10.94	41.90	2857	965,838,488	-74.87	113.9
2	18	0.0	10.53	- 47.42	3179	902,918,120	-75.32	114.1
2	24	0.0	10.05	-136.58	3552	832,640,712	-75.66	114.3
3	6	0.0	9.47	134.48	3997	753,761,432	-75.87	114.5
3	12	0.0	8.73	45.85	4550	664,463,664	-75.91	114.8
3	18	0.0	7.73	- 42.24	5285	561,887,808	-75.68	115.1
3	24	0.0	6.24	-129.32	6370	440,968,032	-74.96	115.5
4	6	0.0	3.41	146.27	8366	290,592,864	-72.95	115.9
4	9	0.0	0.35	107.38	10,481	195,226,352	-70.29	116.2
4	12	0.0	- 9.76	83.36	17,269	67,440,025	-59.95	114.4
4	13	0.0	-26.15	131.27	31,159	7,492,774	-30.31	92.9
4	13	10.5	-19.16	-179.46	36,043	424,795	- 6.69	71.7
4	13	10.6	-18.97	-178.91	36,065	400,000	- 6.40	71.5

1) INERTIAL QUANTITIES

2) CSM BURNOUT, TRANSEARTH INJECTION

[redacted]

6.0 References

(1) Saturn IB SA-204 and Subsequent Design Trajectories, MSFC Memo, R-AERO-DAP-17-65, dated 8 February 1965.

(2) Saturn V Aerodynamic Data, MSFC Memo, M-AERO-A-101-63, dated 23 September 1963.

(3) Saturn V Data Sheets supplied by MSFC under heading "2.4 Ground Rules".

(4) Control Weights Requirements, 16 September 1964, Office of Manned Space Flight, SE-007.000-1.

(5) Lunar Orbit Survey Missions Study, TRW Report 4226-6002-RW000, 1 April 1965.

(6) Extended Stay Lunar Exploration Mission Study, TRW Report 4226-6002,RW000, 7 May 1965.